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PREDICTION TECHNIQUES FOR PRESSURE AND HEAT-TRANSFER DISTRIBUTIONS OVER BODIES OF REVOLUTION IN HIGH-SUBSONIC TO LOW-SUPERSONIC FLIGHT

by

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for the

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ABSTRACT. A literature survey and an evaluation of applicable literature concerning pressure and heat-transfer distributions over bodies of revolution in high-subsonic to low-supersonic flight is presented. Methods for determining the pressure distribution on bodies of revolution are discussed and compared with experimental data. Variations of the modified Newtonian method may be applied to a large class of bodies of revolution for free-stream Mach numbers greater than 0.75.

Appropriate experimental laminar and turbulent heat-transfer data for a blunt-body vehicle at flight Mach numbers between 1.6 and 5 are compared with calculated values. Experimental data could not be found for blunt-body turbulent heat transfer in the free-stream Mach number range from 0.7 to 1.6.

An extensive bibliography covering prediction techniques and experimental data on pressure and heat-transfer distributions is included as the appendix.

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FOREWORD

This report documents an effort to determine engineering methods for the calculation of pressure and heat-transfer coefficient distributions that are both convenient in application and accurate in their predictions as exemplified by agreement with the available experimental data.

The work was performed at the University of Utah during the period from August 1966 to April 1968. This study was made under Contract N60530-67-C-0046 with the Naval Weapons Center, China Lake, Calif. Leo D. Schultz was the technical coordinator, and funds were provided through WepTask RMGA-81-039/216/F009-10-01.

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CONTENTS

NOTATION

B	Velocity gradient at the stagnation point
c_f	Local skin-friction coefficient, $2\tau_w/\rho_e u_e^2$
C_H	Local Stanton number, $\dot{q}_w/\rho_e u_e \left(h_{aw} - h_w\right)$
C_H^*	Local Stanton number based on reference conditions, $\dot{q}_w/\rho^* u_e\left(h_{aw} - h_w\right)$
C_p	Pressure coefficient = $\left(P - P_\infty\right)/(\gamma/2)\ P_\infty M_\infty^2$
c_p	Specific heat at constant pressure
d	Body diameter at base of nose section
F	A function, defined by context
h	Enthalpy
$\bar{h}$	h/h_{s_e}
h_{aw}	Adiabatic wall enthalpy
K	$M_\infty(d/L)$
k	Thermal conductivity
L	Length of nose section
ℓ	Length of body
M	Mach number
Nu	Local Nusselt number based on reference conditions, $\dot{q}_w\left(c_p^*\right) s/k^*\left(h_{aw} - h_w\right)$
P	Pressure at the surface of the body
Pr	Prandtl number, 0.71, $\mu c_p/k$

$\dot{q}$	Heat transfer per unit area per unit time
R	Radius of body cross section normal to the axis of symmetry
$\tilde{R}$	$Re\ \sigma_{s_e}^{1/2}$
Re	Reynolds number, $\rho_{s_e} u_\infty R_o/\mu_{s_e}$
Re^*	Local Reynolds number based on reference conditions, $\rho^* u_e s/\mu^*$
R_o	Hemisphere radius
$\bar{r}$	r_o/R_o
r_b	Body radius at base of nose section
r_c	Radius of blunting at tip of nose section
r_o	Radial distance from axis of symmetry
S	Area of cross section normal to axis of symmetry
s	Wetted length from stagnation point
$\bar{s}$	$\dfrac{s}{R_o}$
T	Temperature
u	Velocity
$\bar{u}$	u/u_e
u_1	Perturbation velocity, x component
u_2	Perturbation velocity, y component
u_3	Perturbation velocity, z component
x	Axial distance from nose

γ	Ratio of specific heats
θ	Angle between flight direction and radius vector from center curvature of nose
θ_b	Local slope of body surface relative to the axis of symmetry (the angle between tangent to body surface and the axis of symmetry)
μ	Viscosity
$\bar{\mu}$	μ_e/μ_{s_e}
ν	Prandtl-Meyer angle
ρ	Mass density
$\bar{\rho}$	ρ_e/ρ_{s_e}
σ_{s_e}	$P_s/\rho_{s_e} h_{s_e}$
τ	Characteristic body parameter, d/ℓ
τ_w	Wall shearing stress
Φ	Velocity potential
ϕ	First-order perturbation velocity potential
$\emptyset$	Second-order perturbation velocity potential

Subscripts

e	Conditions external to the boundary layer
N	Conditions at nose of pointed body
s	Stagnation conditions
t	Total conditions
w	Wall conditions
∞	Free-stream conditions
2	Conditions evaluated behind a shock wave

Superscripts

*	Denotes conditions evaluated at the state corresponding to the reference enthalpy

Section 1.

INTRODUCTION

The purpose of this report is to present a review of the available methods for determining the pressure distribution on bodies of revolution in the subsonic, transonic, and supersonic flight regimes and a comparison of the results of these methods with available experimental pressure distributions. In Section 2, the methods discussed are those that are most useful as "engineering" tools and provide the pressure distribution with a minimum of effort. These methods are

1. Modified Newtonian theory:

   a. An empirically modified approach applicable in the range $M_\infty = 0.75$ to $M_\infty = 2.0$

   b. A hybrid method combining the modified Newtonian and shock-expansion theories for the range $M_\infty > 1.5$

2. Perturbation theory as developed by Van Dyke and Sprieter

3. Similarity rules.

Methods 2 and 3 may be applied over the entire range of Mach numbers. However, these methods require considerably more effort than does Method 1.

In addition, brief reference is made to the most successful of computer techniques used for determining pressure distribution for blunt bodies in supersonic flow.

Section 3 presents a review of the engineering methods of analysis for the convective heating of blunt-body vehicles in the high-subsonic to low-supersonic flight regimes. Comparison of these simplified analytical methods with appropriate experimental data is also presented.

The objective of Section 3 is to present methods of prediction that are both convenient in their utilization and accurate in their agreement with available experimental data. These methods of prediction are (1) of convective heat transfer at the stagnation point of a blunt-body vehicle, and (2) of distribution of the heat transfer over the surface of the vehicle. With this objective in mind, a literature search was performed to locate both the methods of analysis and the appropriate experimental data.

Considerable effort has been devoted to determining the pressure distribution over bodies of revolution in subsonic and supersonic flight. With the exception of a few notable early efforts, the bulk of the work has been done during the past 20 years. The appendix of this report presents a bibliography covering both theoretical developments and available experimental pressure distribution data. This bibliography comprises 567 entries.

Section 2

PRESSURE DISTRIBUTION ON BODIES OF REVOLUTION

MODIFIED NEWTONIAN THEORY

The central hypothesis of Newtonian flow theory is that the component of momentum of the oncoming flow normal to the body surface is transferred to the body while the tangential component remains unchanged. Various authors (Ref. 1) have pointed out that Newtonian theory is reached only through a limiting process as $M_\infty \to \infty$ and $\gamma \to 1$. Cole (Ref. 2) presents a more detailed examination of the relationship between Newtonian theory and gas dynamics; but again, he concludes that Newtonian theory is applicable only for large Mach numbers. Based on these conclusions, Newtonian theory was considered only as a limiting case and was originally used only to describe pressure distributions in the hypersonic flight regime (Ref. 1, 2, 3). However, it was soon found from experimental data that in a modified form, application was valid also in the supersonic flight regime (Ref. 4-9).

In its original form the Newtonian concept provided a coefficient of pressure in the form

$$C_p = 2 \sin^2\theta_b \tag{1}$$

This relationship indicates dependence only on the local slope of the surface, θ_b. Agreement with experiment was not achieved, however, until Lees (Ref. 3) suggested a very useful modification of this expression in the form

$$C_p = C_{p_{max}} \sin^2\theta_b \tag{2}$$

In Eq. 2, $C_{p_{max}}$ is the maximum value of the coefficient of pressure on the body surface. For a blunt body in supersonic flow, the maximum value of C_p is at the stagnation point, and its value may be conveniently computed from the following expression:

$$C_{p_{max}} = \frac{\left(P_{t_2}/P_\infty\right)-1}{(\gamma/2)\, M_\infty^{\,2}} \tag{3}$$

where P_{t_2}/P_∞ is determined from the Rayleigh formula:

$$P_{t_2}/P_\infty = \left[\frac{(\gamma+1)M_\infty^{\ 2}}{2}\right]^{\gamma/\gamma-1} \left[\frac{\gamma + 1}{2\gamma M_\infty^{\ 2} - (\gamma-1)}\right]^{1/\gamma-1} \tag{4}$$

Values of $C_{p_{max}}$ computed by using Eq. 3 and 4 are compared with experimental values in Fig. 1. Figure 1 demonstrates an excellent agreement between theory and experiment. As M_∞ becomes large, computed values of $C_{p_{max}}$ for air approach 1.83 rather than 2.

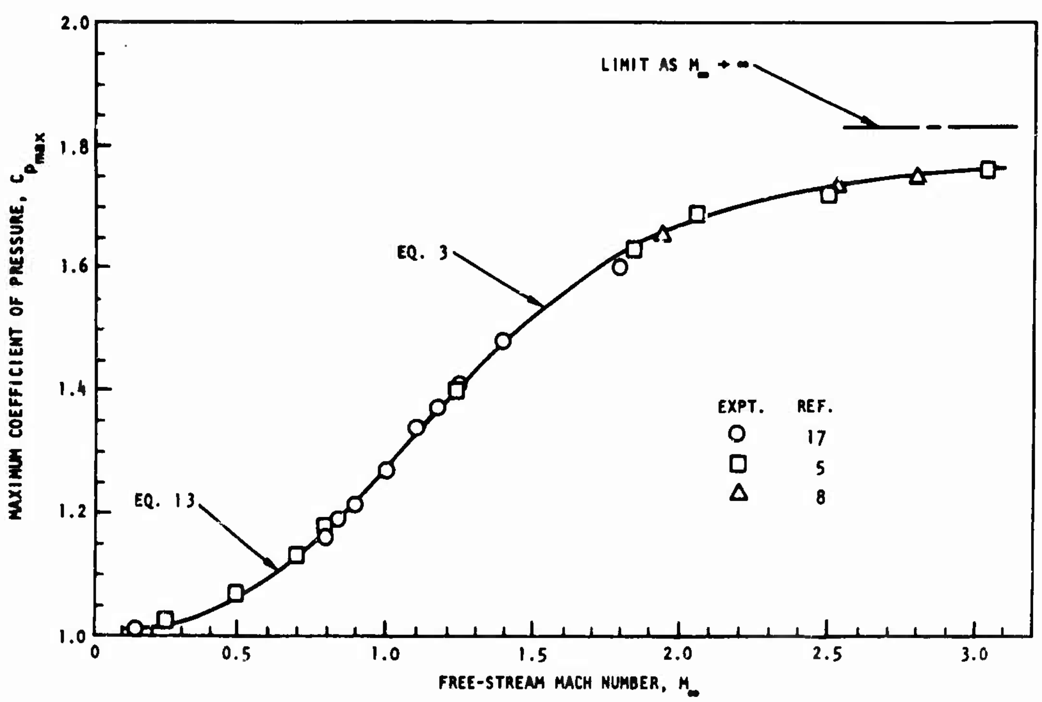


FIG. 1. Maximum Coefficient of Pressure for Blunt Bodies of Revolution as a Function of Mach Number.

Oliver (Ref. 10) presents the following simple formula which may be used for determining $C_{p_{max}}$ if $M_\infty \geq 2$:

$$C_{p_{max}} = \frac{\gamma+3}{\gamma+1}\left[1 - \frac{2}{\gamma+3}\,\frac{1}{M_\infty^2}\right] \tag{5}$$

From Eq. 2, 3, 4, and 5, C_p is seen to be a function of M_∞ and γ, as well as of θ_b. Comparison of Eq. 2 with experimental data for hemispherically capped cylinders is presented in Fig. 2. The Mach numbers of the data ranged from 1.97 to 4.76. Equation 2 represents the data quite adequately.

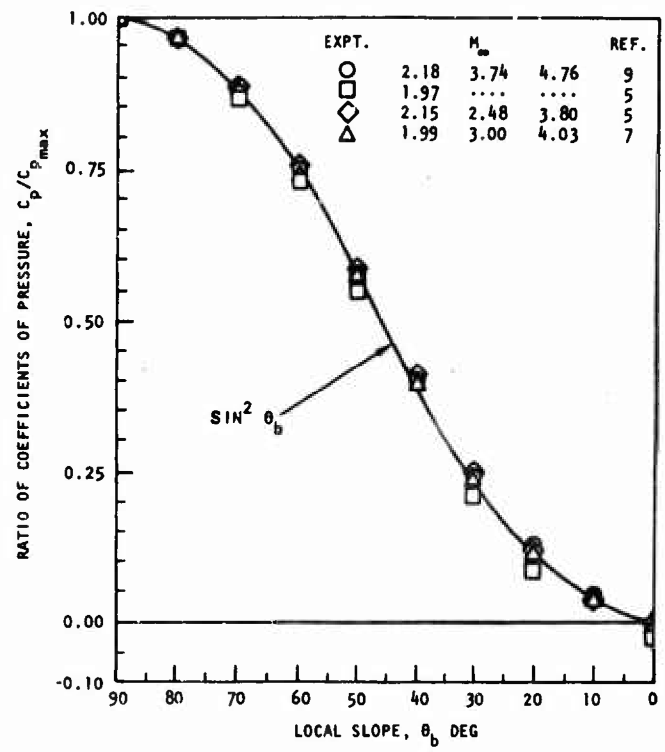


FIG. 2. Distribution of Pressure Coefficient Ratio Over a Hemispherical Nose, $M_\infty \geq 1.97$.

For bodies of revolution which are only slightly blunted or for a pointed body, the modified Newtonian method may also be applied if $C_{p_{max}}$ is computed as suggested by Love (Ref. 6):

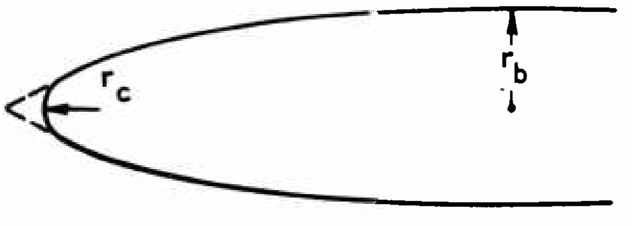
$$C_{p_{max}} = C_{p_N} / \sin^2 \theta_N \tag{6}$$

where

$$C_{p_N} = C_p$$

from tables of conical flow (Ref. 11) for a cone semivertex angle, θ_N, corresponding to the local slope at the nose of the pointed body.

An example of what is meant by an "only slightly blunted" body would be a hemispherically capped ogive with a cap radius that is small relative to the maximum cross-section radius of the ogive (say r_c/r_b less than approximately 0.2). For such a case, the C_p at the stagnation point will correspond to that of Eq. 3, but the distribution away from the stagnation point must be determined on the basis of the $C_{p_{max}}$ given by Eq. 6 for an equivalent pointed body (Ref. 12). By way of explanation, an equivalent pointed body is that pointed body which would exist if the nose were not blunted.



Love (Ref. 6) presents a very interesting comparison of this generalized Newtonian theory (Eq. 2 and 6) with experimental data for a variety of bodies of revolution over Mach number ranges from 2 to 7. This comparison is reproduced in Fig. 3 with the scatter in the data indicated by the band width.

The modified or generalized Newtonian prediction begins to deviate from experimental values as the local slope approaches zero, or as a point of discontinuity in slope is approached. Lees and Kubota (Ref. 13) suggest that in such a region, a local shock-expansion method be used to predict the distribution. To accomplish this the modified Newtonian equation

$$C_p = C_{p_{max}} \sin^2 \theta_b$$

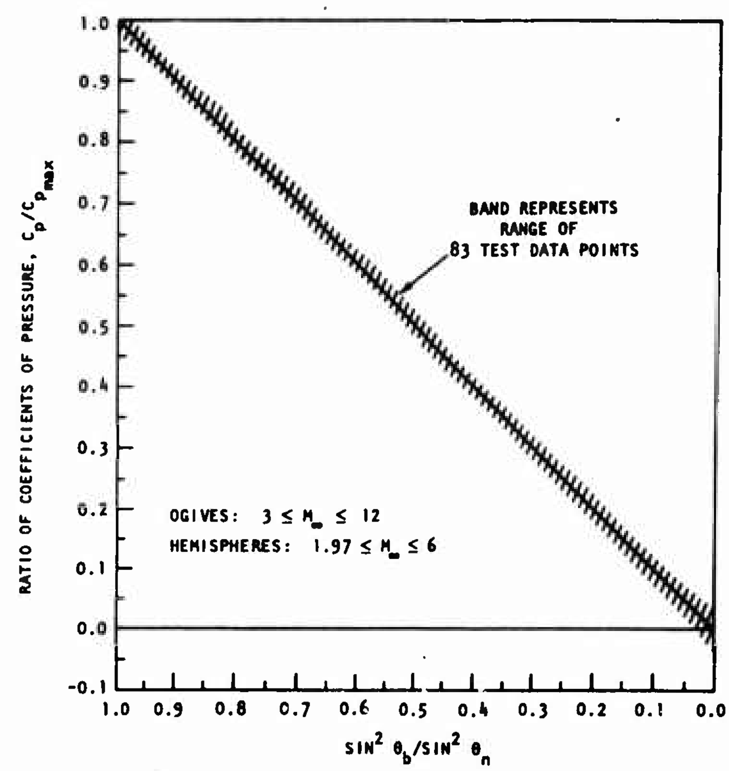


FIG. 3. Love's Correlation for Generalized Newtonian Theory.

and the Prandtl-Meyer relation

$$\frac{1}{P}\left(\frac{dP}{d\theta_b}\right) = \gamma M^2/\sqrt{M^2 - 1} \tag{7}$$

are matched at the point on the nose surface where both the pressure and pressure gradient, predicted by these two solutions, are equal. This method was further investigated by Wagner (Ref. 14), Jones (Ref. 15), and Blick and Francis (Ref. 16). Blick and Francis suggest the following formula for determining the matching angle in the Mach number range 2 to 15:

$$\theta_m = 0.349 + \left[0.198 + 0.80\ (M_\infty - 2.8)^{-0.455}\right] \cos^{-1}\left[1.05/(\ln M_\infty)\right] \tag{8}$$

Vendemia (Ref. 9) presents an excellent summary of this hybrid modified Newtonian/shock-expansion method. A number of comparisons of the method with experimental data or "exact" theoretical methods for ogives, Von Karman minimum drag, and 3/4 power law bodies in the Mach range $M_\infty = 1.5$ to 6 and for hemispheres in the $M_\infty = 1.82$ to 4.76 range are presented in Ref. 9. Two of these comparisons are reproduced as Fig. 4 and 5. These figures show that the agreement between the present theory and experiment or the more "exact" theory is excellent.

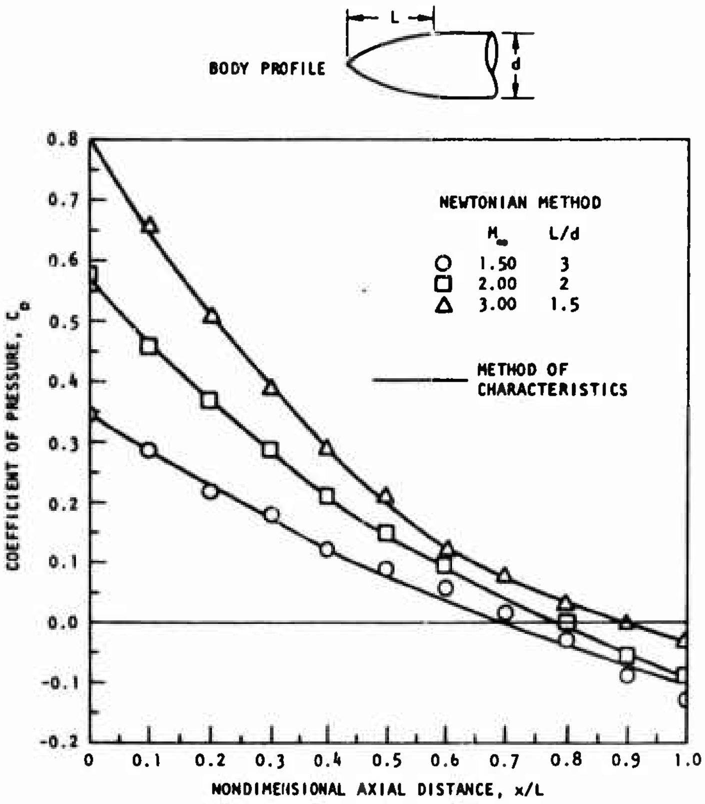


FIG. 4. Comparison of Modified Newtonian/Shock-Expansion Method With Method of Characteristic Solution for Tangent Ogive Bodies at Supersonic Mach Numbers.

Vendemia also developed the following expression to determine the matching point for the modified Newtonian and shock-expansion methods. From

$$C_p = C_{p_{max}} \sin^2\theta_b = \frac{P - P_\infty}{(\gamma/2)P_\infty M_\infty{}^2}$$

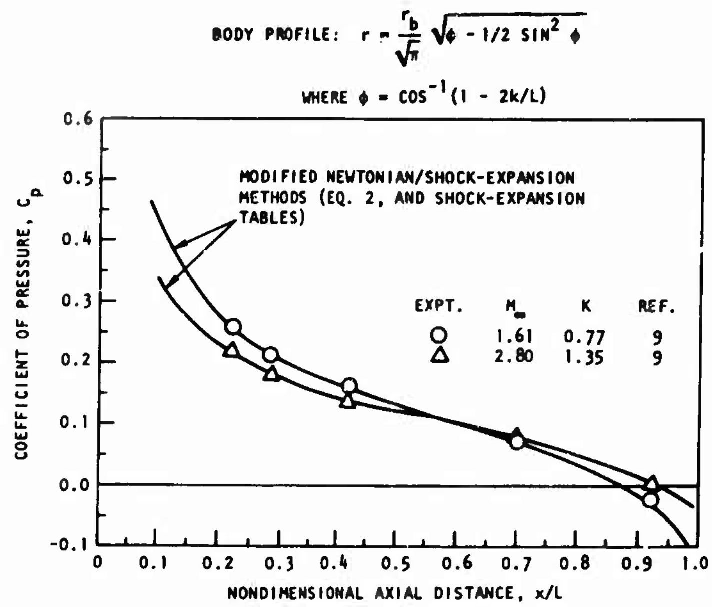


FIG. 5. Comparison of Modified Newtonian/ Shock-Expansion Method With Experimental Data for Von Karman Minimum Drag Body.

it follows that

$$\frac{1}{P}\frac{dP}{d\theta_b} = \frac{(\gamma/2)\, M_\infty^{\,2} C_{p_{max}} \sin 2\theta_b}{1 + (\gamma/2)\, M_\infty^{\,2}\, C_{p_{max}} \sin^2\theta_b} \tag{9}$$

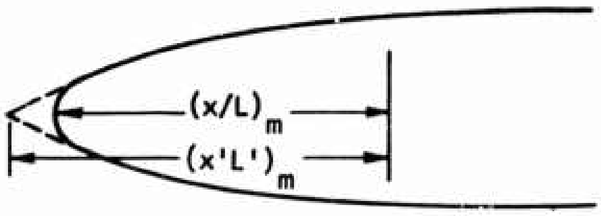
Computations are then made to determine the matching point θ_m where P and $dP/d\theta_b$ are the same from both equations. Vendemia then presented the results in graphical form as a plot of the matching angle θ_m versus M_∞ for hemispheres and hemisphere cones. The result is reproduced in Fig. 6. It should be noted that for low free-stream Mach numbers, there is no point at which both the pressure and pressure gradient match, for the two methods and the curve of Fig. 6 have been extrapolated in this area. Vendemia (Ref. 9) found that the matching point on a tangent ogive can be determined from the following empirical equation:

$$(x'/L')_m = 0.60 + 0.15K \text{ for } K \leq 2.667 \tag{10}$$

where

$$K = M_\infty (d/L)$$

This equation may also be applied to slightly blunted ogives by relating the $(x'/L')_m$ computed from Eq. 10, which would be the nondimensional distance from the nose of the equivalent pointed ogive to the corresponding point on the blunted ogive.



For blunted ogives where r_c/r_b is considerably greater than 0.2, an alternative method of determining the pressure distribution, to be discussed shortly, will be more effective.

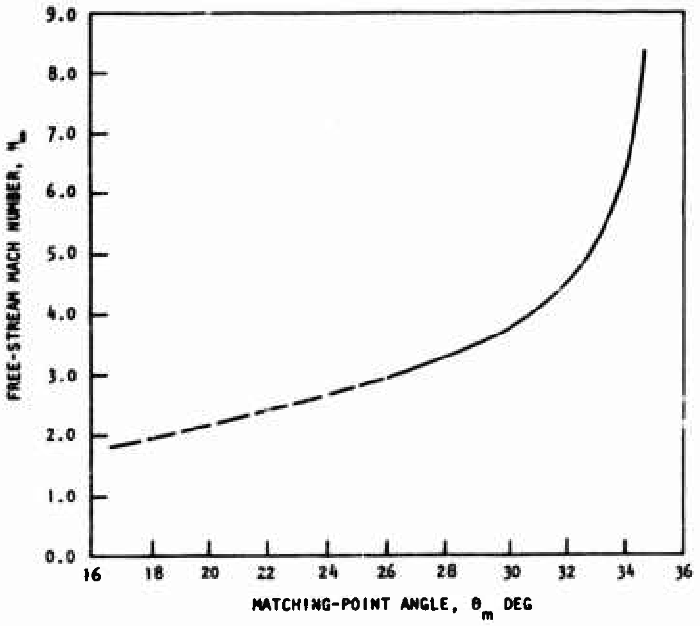


FIG. 6. Matching-Point Angle for Hemisphere or Hemisphere-Cone Body. (Vendemia graph, Ref. 9.)

For predicting the pressure over a cylindrical afterbody, Vendemia (Ref. 9) gives the following equation:

$$C_p = C_{p_{(x/L=1)}} \exp\left(-\frac{\Delta x/L}{K}\right) \tag{11}$$

where $\Delta x/L$ is the nondimensional distance from the base of the nose

where

$x/L = 1$

The method outlined above, using Eq. 2, 6, 10, and 11, has been applied to a hemispherically capped ogive body and is compared with experimental data (Ref. 12) in Fig. 7 and 8. These figures illustrate the usefulness of this method in obtaining good results. It should also be noted that Eq. 3 again gives the correct value of C_p at the stagnation point. The calculations for Fig. 8 are given as Example 1, page 17.

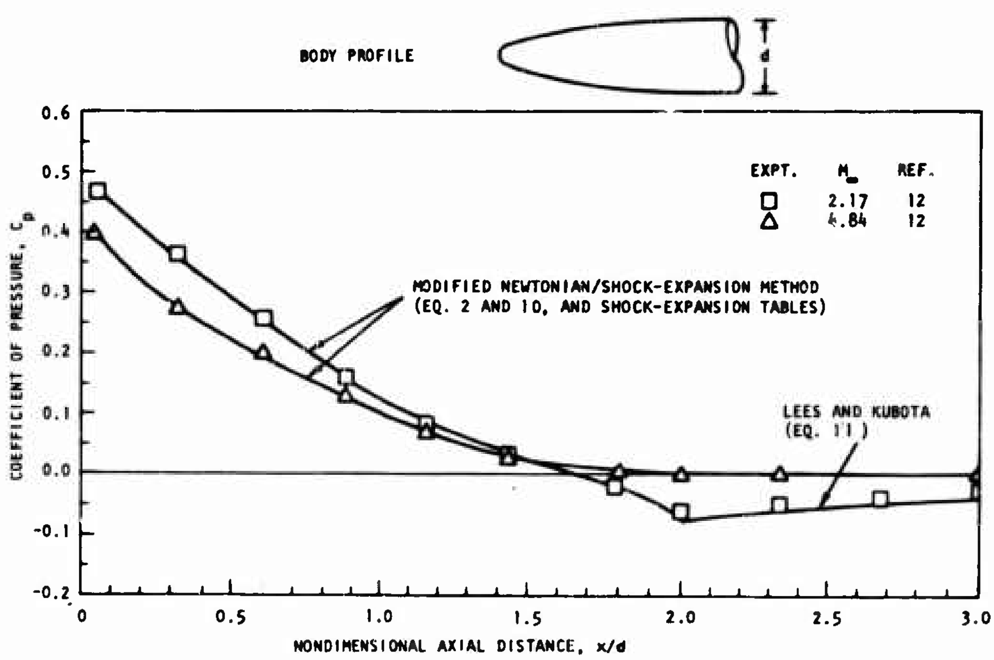


FIG. 7. Pressure Distribution on a Spherically Tipped Tangent Ogive Body.

A further significant application of modified Newtonian theory was carried out by Andrews (Ref. 8). For the case of a hemisphere cylinder, an empirical modifying function was added to the modified Newtonian

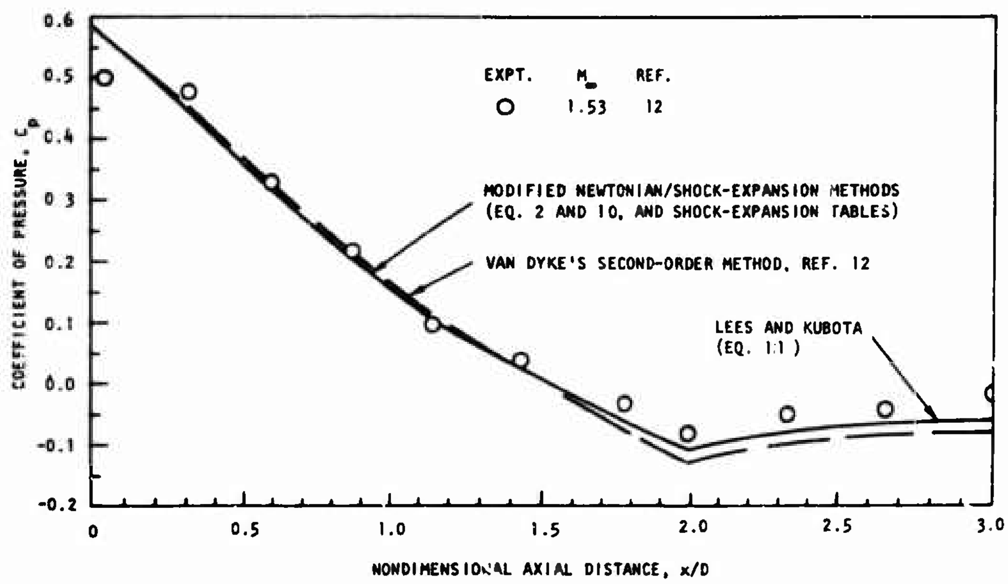


FIG. 8. Comparison of Modified Newtonian/Shock-Expansion and Van Dyke's Second-Order Method With Experimental Data for a Spherically Tipped Tangent Ogive Body.

expression, Eq. 2, and agreement was achieved with experimental data from Ref. 8 and 17 for a Mach number range from 0.75 to 2.0. The equation used was

$$C_p = C_{p_{max}} \sin^2 \theta_b + F\left(\theta_b, M\right) \tag{12}$$

where

$$F\left(\theta_b, M\right) = \left\{0.78\ M_\infty^{-2.27} \sin \theta_b - 0.95 \exp\left[-2.235\ (M_\infty - 1)\right]\right\} \cos \theta_b$$

This equation is presented in Fig. 9 and 10 and compared with experimental results (Ref. 5, 8, 9, and 17).

In the subsonic regime, $C_{p_{max}}$ for blunt bodies is computed from the isentropic relation

$$C_{p_{max}} = \frac{\left(1 + \frac{\gamma - 1}{2} M_\infty^2\right)^{\gamma/\gamma - 1} - 1}{(\gamma/2)\ M_\infty^2} \tag{13}$$

This equation is compared with experimental results in Fig. 1.

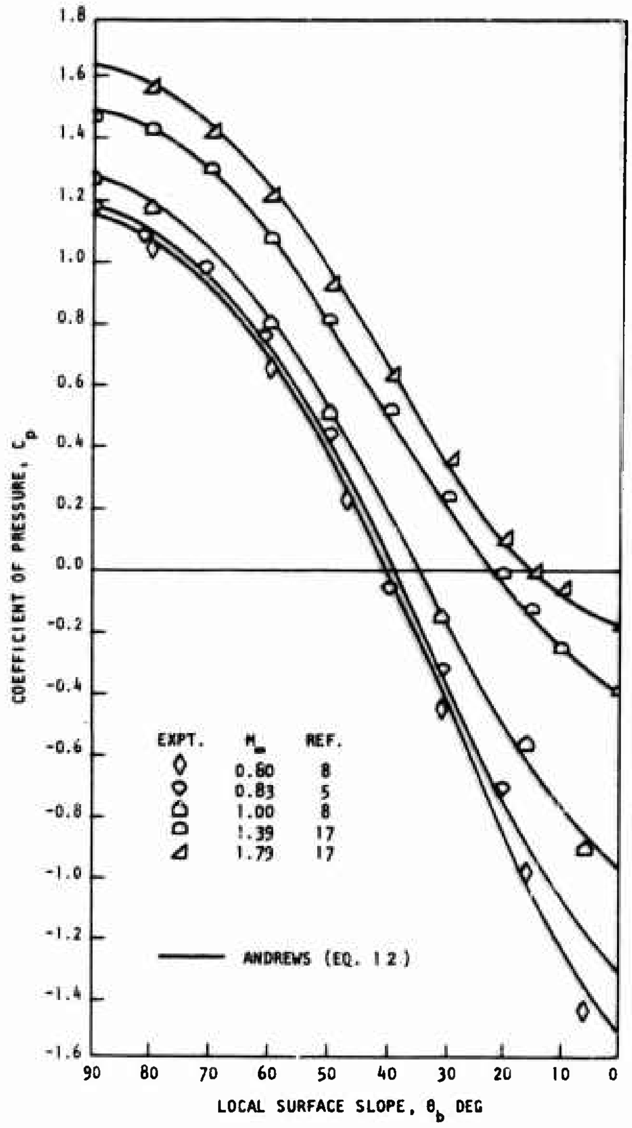


FIG. 9. Comparison of Eq. 12 With Experimental Data for Hemisphere Cylinders.

A modified form of Eq. 11 may also be used in combination with Eq. 2 or 12 to predict the pressure distribution on a hemisphere cone combination in the supersonic regime. The pressure distribution for the hemispherical nose is determined by using either Eq. 2 or 12 with Eq. 3. The cone tables (Ref. 11) are then used to determine the pressure coefficient for a cone, C_{p_c}, with a semivertex angle, θ_N, equal to the slope

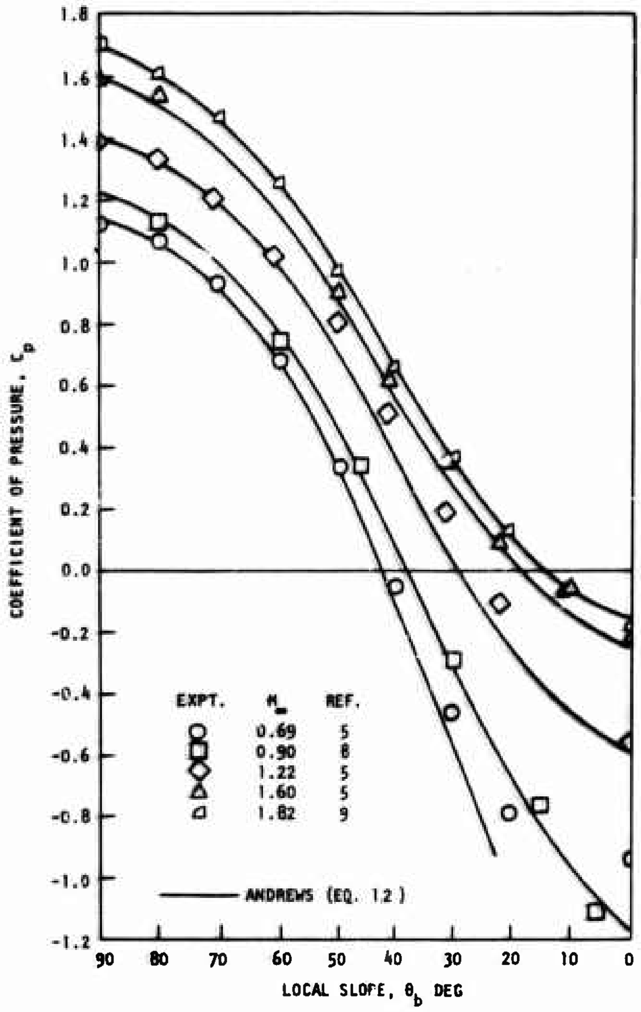


FIG. 10. Comparison of Eq. 12 With Experimental Data for Hemisphere Cylinders.

of the conical section of the body of interest. This will be the value of C_{p_c} reached on the cone surface some distance downstream of the hemisphere-cone junction. The pressure distribution on the conical section can then be approximated:

$$C_p = \Delta C_{p_{(j-c)}} \exp\left(- \frac{\Delta s/\ell}{K}\right) \tag{14}$$

where

$$\Delta C_{p_{(j-c)}} = C_{p_j} - C_{p_c}$$

$$C_{p_j} = C_p \text{ from Eq. 2 at the hemisphere-cone junction}$$

$$C_{p_c} = C_{p_c} \text{ for a cone of } \theta_N$$

The use of this approach is illustrated by comparison with experimental data (Ref. 18) in Fig. 11 and 12.

For a blunt body with other than a hemispherical nose, Eq. 12 may be used with good results if the function F is multiplied by the fineness parameter (r/L):

$$C_p = C_{p_{max}} \sin^2 \theta_b + (r_b/L)\, F(\theta_b, M) \tag{15}$$

r_b = radius at the base of the nose section

L = axial distance from stagnation point to nose base

$F(\theta_b, M)$ = same as that in Eq. 12

Eq. 15 indicates that the influence of the modifying function $F(\theta_b, M_\infty)$ decreases as the body becomes more slender.

This equation was used to compute the pressure distribution on a body having an elliptical nose with $r_b/L = 0.61$. The results are compared with experimental data (Ref. 19) in Fig. 13. Equation 15 was also used to predict the distribution on a prolate spheroid (ellipsoid of revolution) with $r_b/L = 0.167$, and the results are compared with experimental data (Ref. 20) in Fig. 14. From Fig. 14, note that the pressure gradient is predicted quite well, even though the actual

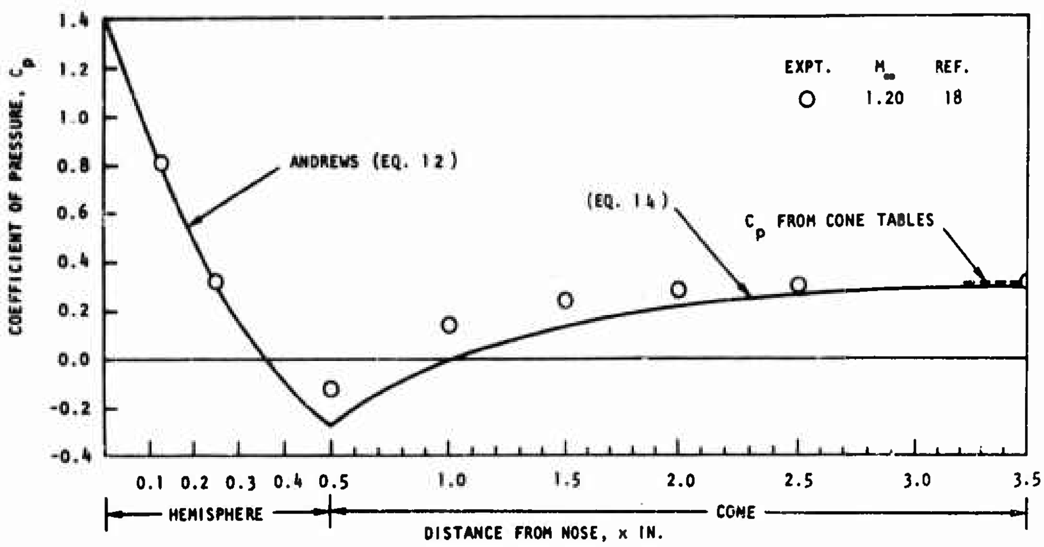


FIG. 11. Pressure Distribution on a Hemispherically Capped 15-Deg Cone.

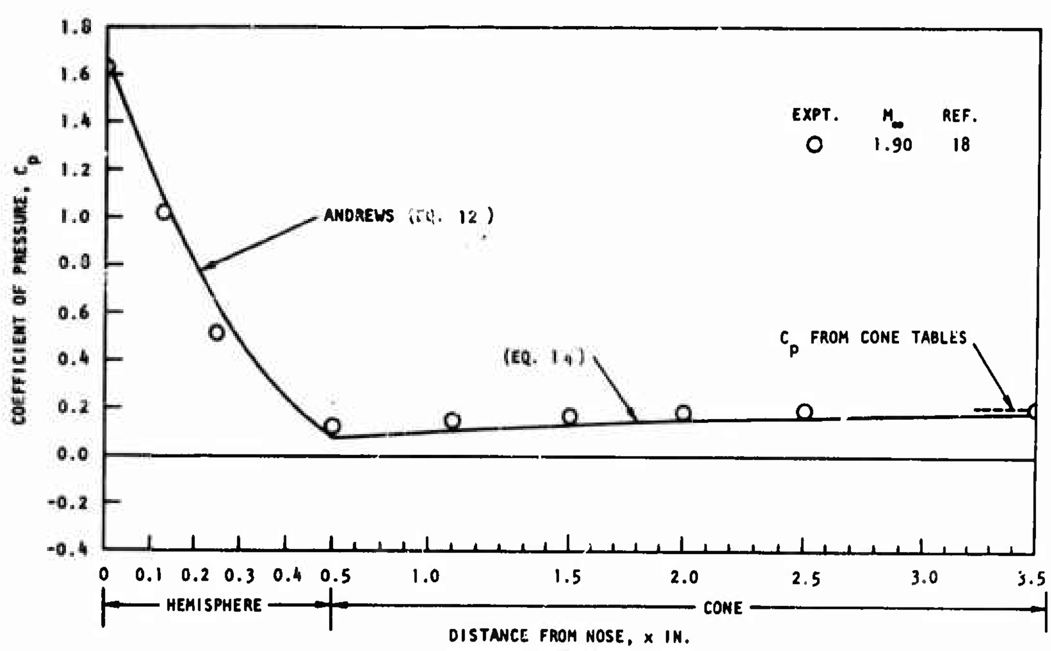


FIG. 12. Pressure Distribution on a Hemispherically Capped 15-Deg Cone.

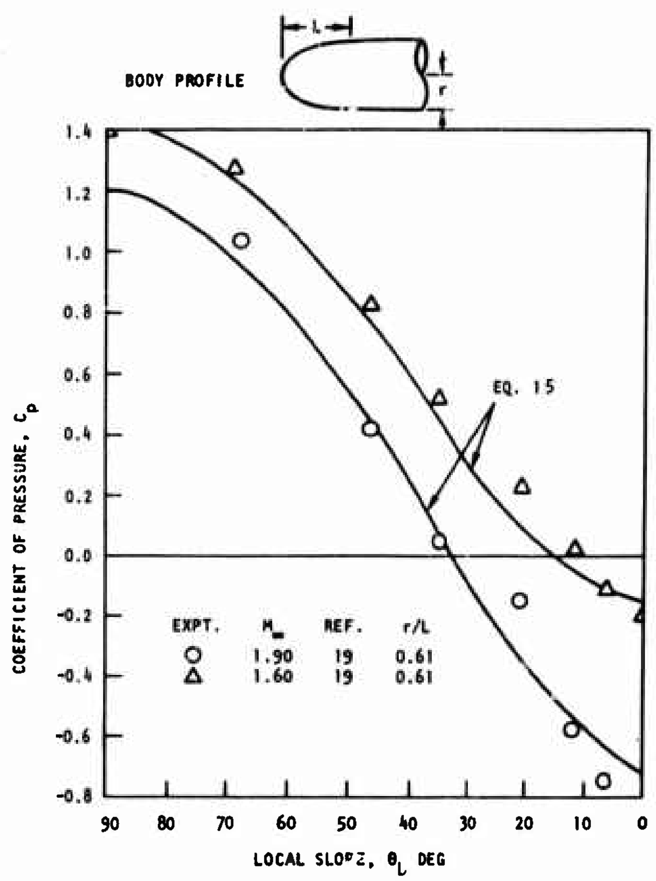


FIG. 13. Pressure Distribution on an Ellipsoid of Revolution-Nosed Cylinder.

magnitude of C_p from Eq. 15 and experiments differ somewhat. As an example of this approach, the calculations corresponding to Fig. 13 are presented on page 27.

The chief advantage of the Newtonian method is that with a minimum of effort, good accuracy is available.

Table 1 presents a summary of the available pressure distribution data.

EXAMPLE 1: MODIFIED NEWTONIAN/SHOCK-EXPANSION METHOD

As a numerical example of the modified Newtonian/shock-expansion method, consider again the data shown in Fig. 7 for $M_\infty = 2.17$. The body

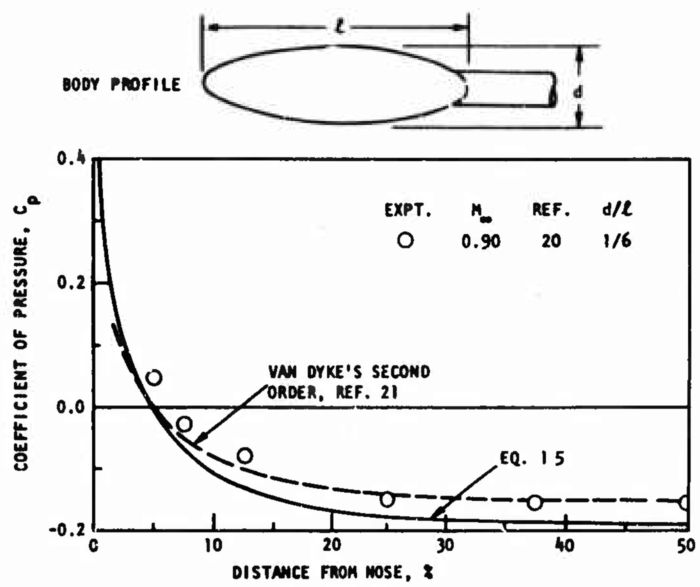


FIG. 14. Comparison of Prediction Techniques With Experiment Pressure Distribution on a Prolate Spheroid. (Data from Ref. 24.)

is a spherically capped ogive with $r_c/r_b = 0.158$. By assuming that this body may be treated in terms of an equivalent pointed body, the pressure distribution may be computed as follows:

1. The coefficient of the pressure at the stagnation point (Fig. 1) is given as $C_{p_c} = 1.70$.

2. To determine the distribution away from the stagnation region, first determine θ_N for the equivalent pointed body. For an ogive,

$$\sin \theta = \frac{L - x}{R}$$

where

L = length of ogive

x = axial distance from the nose

R_c = radius of ogive surface curvature (in the x-y plane)

TABLE 1. Pressure-Distribution Data

Data Source	Body Configuration	Mach Number	Test Reynolds Number/ft	Theoretical Comparison
Andrews, J. S. (AD 465-140, Boeing D2-22947-1; 1964)	Hemisphere cylinder	0.8 0.9 1.0 1.076	$5.5 \cdot 10^6$ $6.0 \cdot 10^6$ $6.1 \cdot 10^6$ $6.2 \cdot 10^6$	Equation 12.
(Reported in Andrews but original source is Morris and Grobli (16))		1.39 1.79	Not reported	
Baer, A. L. (AD 261-501, AEDC TN 61-96; 1961)	Hemisphere cylinder	1.99 3.00 4.03 5.06 6.03 8.10	$2.02 \cdot 10^6$ $4.2 \cdot 10^6$ $5.3 \cdot 10^6$ $6.1 \cdot 10^6$ $4.4 \cdot 10^6$ $2.5 \cdot 10^6$	Modified Newtonian Equation
Buglia, J. J. (NASA TN D955; 1961)	Hemisphere cone $\theta_{cone} = 14°30'$	Free flight 1.97 to 3.14	Approximately $10 \cdot 10^6$	Modified Newtonian
Carter, H. S. and Bressette, W. E. (NACA RM L57C18; 1957)	Hemisphere cone Flat-Faced cone	2.0 2.0	$14 \cdot 10^6$ $14 \cdot 10^6$	None
Chauvin, L. T. (NACA RM L52K06; 1952)	Hemisphere cone	2.05 2.54 3.04	$4.4 \cdot 10^6$ $4.57 \cdot 10^6$ $4.16 \cdot 10^6$	None
Estabrook, B. B. (AEDC TR-59-12, AD 216 698)	20° Cone Cylinder and parabolic arc-cylinder	0.70 0.80 0.90 0.95 0.975 1.00 1.025 1.050 1.10 1.15 1.20 1.30 1.40	Range--from 1.36×10^6 to 4.0×10^6	Compared with theoretical cone data
Gardiner, G. K. (AD 461 744, Litton Systems, HADES-12-65-1; 1965)	Hemisphere cone ($r_c/r_b \simeq 0.07$)	0.3 0.5 0.65 0.8 0.9 1.0 1.1 1.2 1.3	2×10^6 3.14×10^6 3.3×10^6 4.15×10^6 4.30×10^6 4.40×10^6 4.40×10^6 4.38×10^6 4.30×10^6	None
Graham, F. J. and Butler, C. B. (AD 240 084, AEDC TN 60-128; 1960)	Hemisphere cone cylinder and ellipsoide cylinder	0.50 0.80 0.85 0.90 0.95 0.975 1.00 1.05 1.10 1.20 1.60	Range of Re $4.0 \cdot 10^6$ to $7.2 \cdot 10^6$	None
Katz, J. R. (AD 159 801, NAVORD R5849; 1958)	Hemisphere cylinder	0.26 0.49 0.69 0.83 1.22 1.6 2.15 2.48 4.20	(Re not reported)	Modified Newtonian
	Hemisphere cone	3.12 5.64 5.80		

TABLE 1. (Continued)

Data Source	Body Configuration	Mach Number	Test Reynolds Number/ft	Theoretical Comparison
Lord, D. R. and Ulman, E. F. (NACA RM L51L20; 1952)	Ogive Cylinder	4.04		Perturbation theory and method of characteristics
Love, E. S. (Journal of Aerospace Sciences, May 1959)	Hemisphere	1.97 3.8 5.8 6.9	(Re not reported)	Modified Newtonian Equation
	Cone $\theta_n = 45°$	6.9		Generalized Newtonian Equation
	Ogive [Md/L = 4] and [h(d/L) = 0.5] Cone $\theta_n = 41°$			Exact theoretical methods compared with Generalized Newtonian Theory Equation
Matthews, C. W. (NACA TN 2519, 1954	Prolate spheroid $\ell/d = 6,10$	0.3 0.6 0.8 0.9 0.92 0.94	$2.7 \cdot 10^6$ to $3.95 \cdot 10^6$	Prandtl-Glauret correction to incompressible potential flow solution
Nichols, J. H. (AD 225 362 362, AEDC-TN-59-100)	8.5 diameter and 10.0 diameter cone cylinder and 8.5 diameter ogive cylinder	0.90 0.95 1.00 1.05 1.10 1.20 1.30 1.40 1.50 1.60	approximately 4.10^6	None
Okauchi, K. (AD 291 599, NAVWEPS R 8048; 1962) [see also Johanson, V. F., NOTS TN 4061-77; 1961]	Hemispherically-capped ogive ($r_c/r_b = 0.16$)	0.75 0.85 0.95 1.05 1.15	(Re not reported)	None
		1.53 2.17		Van Dyke
		3.24 4.84		Method of Characteristics
Pearson, A. O. (NASA TN D1961; 1963)	Hemisphere cone	0.25 0.40 0.60 0.80 0.85 0.90 0.95 1.00 1.05 1.10 1.15 1.20 1.47 1.60 1.90	3.0×10^6 2.3 3.1 3.[illegible] 3.8 3.9 4.0 4.1 4.1 4.2 4.2 4.2 3.0 3.0 3.0	None
Perkins, E. W., Jorgensen, L. H., and Sommer, S. C. (NACA R 1386; 1958	Hemisphere cylinder	1.97 3.04 3.80	4.10^6 4.10^6 4.10^6	Newtonian Theory Equation 1
	Hemisphere cone, θ_c = 6°59'	1.50 1.97 3.10	$4.1 \cdot 10^6$ $4.14 \cdot 10^6$ $4 \cdot 10^6$	Cone Theory
	Hemisphere cone, θ_c = 5°10'	1.54 1.97 3.06	$4.1 \cdot 10^6$ $4.14 \cdot 10^6$ $4 \cdot 10^6$	
Spreiter, J. R. and Alksne, A. Y. (NASA TR-2; 1959)	Parabolic arc bodies ℓ/d = 6, 10, and $6\sqrt{2}$	0.80 0.85 0.90 1.0	(Re not reported)	Perturbation theory
	Cones	1.0		
Stine, H. A. and Wanlass, K. (NACA TN 3344; 1954)	Hemisphere cylinder	1.97 3.04 3.80	$3 \cdot 10^6$ to $6 \cdot 10^6$ $2.8 \cdot 10^6$ to $4 \cdot 10^6$ $2.84 \cdot 10^6$	Newtonian Theory Equation 1 and Incompressible Theory

From the body geometry, L may be determined from the given values r_b and R_c.

$$\sin \psi = L/R$$

$$\psi = \cos^{-1}\left(\frac{R - rb}{R}\right)$$

If

$$r_b = 0.592 \text{ inch}$$

$$R_c = 5.534 \text{ inches}$$

then

$$\psi = 27 \text{ deg}$$

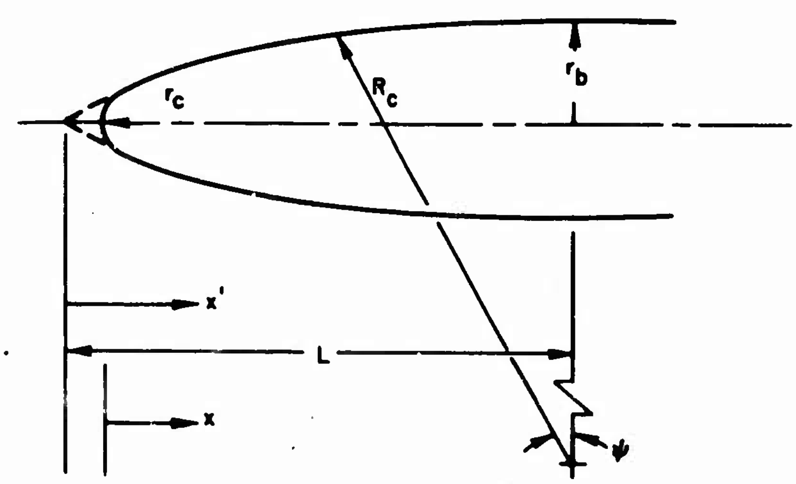
and

$$L = 2.50 \text{ inches}$$

$$\theta_N = \text{local slope at the nose } (x = 0)$$

$$\sin \theta_N = \frac{L - 0}{R} = 0.454$$

$$\theta_N = 27 \text{ deg}$$



3. Values for C_{p_N} and for $\left(P/P_t\right)_N$ may be found in Ref. 11, or in the National Advisory Committee for Aeronautics publication, NACA Report 1135, Washington, D. C., 1953; for example,

$$C_{p_N} = 0.53 \text{ and } \left(P/P_t\right)_N = 0.29$$

From Eq. 6,

$$C_{p_{max}} = C_{p_N}/\sin^2\theta_N = 2.59$$

4. Modified Newtonian theory gives:

x/D	C_p Theory	C_p Experiment	
0	1.70	1.68	(Stagnation value, Fig. 1)
0.046	0.46	0.46	
0.325	0.34	0.36	
0.604	0.24	0.26	
0.883	0.15	0.16	
1.161	0.084	0.085	
1.440	0.037	0.030	
1.719	0.012	-0.03	
1.999	0	-0.08	

One sees that the modified Newtonian theory begins to deviate significantly from experimental data as θ becomes small. This was expected.

5. The next step is to find the point beyond which the shock-expansion method should be used. Equation 10 may be used:

$$(x'/L')_m = 0.60 + 0.15K$$

$$K = M_\infty(d/L) = 2.17\left(\frac{1.184}{2.50}\right)$$

$$K = 1.03$$

$$(x'/L')_m = 0.60 + 0.15(1.03) = 0.75$$

$$x'_m = 1.87$$

$$x_m = 1.74$$

$$(x/d)_m = 1.47$$

6. Compute C_p at the matching point using modified Newtonian theory:

$$C_p = C_{p_{max}} \sin^2\theta_{mp}$$

$$C_p = 2.59 \left(\frac{L' - x'}{R}\right)^2$$

$$= 2.59 \left(\frac{2.50 - 1.87}{5.534}\right)^2$$

$$C_p = 0.033$$

7. From the definition of C_p,

$$C_p = \frac{P - P_\infty}{(\gamma/2)P_\infty M_\infty^{\ 2}} = \frac{P - P_\infty}{q_\infty} = \frac{P}{P_t}\left(\frac{P_t}{q_\infty}\right)_N - \frac{P_\infty}{q_\infty}$$

Substituting

$P_t = P_{t_N}$ along the cone surface

and

$$C_{p_N} = \frac{P_N - P_\infty}{q_\infty} = \frac{\left(P/P_t\right)_N - \left(P_\infty/P_t\right)_N}{\left(q_\infty/P_t\right)_N}$$

one obtains

$$C_p = \frac{P}{P_t}\,\frac{C_{p_N} + P_\infty/q_\infty}{\left(P/P_t\right)_N} - \frac{P_\infty}{q_\infty}$$

and

$$\frac{P}{P_t} = \left(C_p + P_\infty/q_\infty\right)\frac{\left(P/P_t\right)_N}{C_{p_N} + P_\infty/q_\infty}$$

From these equations, the shock-expansion method may be started. For the first increment, use $C_p = 0.033$ at $x/d = 1.47$ (modified Newtonian value).

$$\left(P/P_t\right)_1 = 0.033 + 0.30\left(\frac{0.29}{0.56 + 0.30}\right)$$

as

$$P_\infty/q_\infty = 2/\gamma M^2_\infty = 0.30$$

From a table of Prandtl-Meyer, functions for $P/P_t = 0.112$, $\nu_1 = 28.56$.

8. Compute the C_p at the next station downstream, Station 2, as follows:

$$\Delta\theta = \left|\theta_2 - \theta\right|$$

$$\nu_2 = \nu_1 + \left|3.47 - 6.5\right| = 28.56 + 3.03 = 31.59$$

From the tables,

$$\left(P/P_t\right)_2 = 0.095$$

$$C_{p_2} = \left(P/P_t\right)_2 \frac{C_{p_N} + P_\infty/q_\infty}{\left(P/P_t\right)_N} - P_\infty/q_\infty$$

$$C_{p_2} = (0.095)\ \frac{0.56 + 0.30}{0.29} - 0.30$$

$$C_{p_2} = 0.28 - 0.30 = -0.02$$

Repeat the process for the next station, Station 3:

$$P/P_t = 0.095$$

$$\nu_2 = 31.59$$

$$\nu_2 = 31.59 + \left|\theta_3 - \theta_2\right| = 31.59 + \left|0 - 3.47\right|$$

$$\nu_3 = 35.06$$

$$\left(P/P_t\right)_3 = 0.0763$$

$$C_{p_3} = 0.0763\ (2.96) - 0.30$$

$$C_{p_3} = -0.074$$

9. For the cylindrical portion of the body, use Eq. 11:

$$C_{P_4} = C_{P_3} \exp(-\Delta x/LK)$$

$$\frac{\Delta x}{L} = \frac{\Delta x/p}{L/p} = \frac{2.337 - 1.999}{2.111}$$

$$K = 1.03$$

$$\frac{\Delta x}{LK} = \frac{0.338}{2.111(1.03)} = 0.155$$

$$\exp(0.155) = 1.167$$

$$C_{P_4} = \frac{-0.074}{1.167} = -0.063$$

$$C_{P_5} = -0.074 \exp(-0.658/2.175) = \frac{0.074}{1.353}$$

$$C_{P_5} = -0.054$$

The results of this process may be summarized as follows:

x/D	C_p Calculation	C_p Experiment
0	1.70	1.68
0.046	0.46	0.46
0.325	0.34	0.36
0.604	0.24	0.26
0.883	0.15	0.16
1.161	0.084	0.085
1.440	0.037	0.030
1.47	0.033	.....
1.719	-0.020	-0.025
1.999	-0.074	-0.060
2.337	-0.063	-0.050
2.657	-0.054	-0.040

The calculations and data appear in Fig. 7.

EXAMPLE 2: EQUATION 15

As an example of the use of Eq. 15, consider an eliptically nosed cylinder (see Fig. 13) R/L = 0.61 at M_∞ = 0.90:

$$C_p = C_{p_{max}} \sin^2\theta_b + (R/L)F\left(\theta_b, M_\infty\right)$$

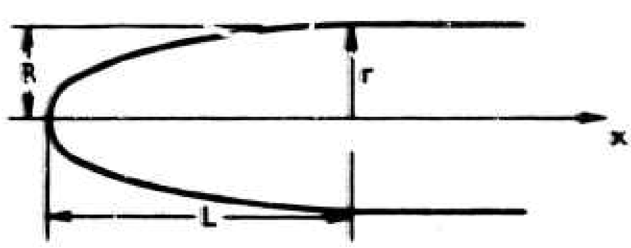
1. From Fig. 1, $C_{p_{max}}$ = 1.215 at M_∞ = 0.9.

2. For these coordinates,

$$\tan\theta = (R^2/L^2)(-x/r)$$

where

$$r = \sqrt{(R^2/L^2)(L^2-x^2)}$$



$-x$	θ_b, deg	$C_{p_{exp}}$
2.684	68	1.075
2.359	45	0.42
2.033	33.5	0.05
1.464	20.7	-0.15
0.935	12.3	-0.58
0.488	6.25	-0.75

3. For convenience in computation, choose calculated points at

θ_b	$\sin^2\theta_b$	$C_{p_{max}} \sin^2\theta_b$	$F\left(\theta_b, M_\infty\right)$	$(R/L)F\left(\theta_b, M_\infty\right)$
90	1.000	1.215	0.0	0
80	0.9698	1.18	-0.037	-0.024
70	0.8830	1.072	-0.088	-0.045
60	0.7500	0.911	-0.165	-0.10
50	0.5868	0.714	-0.276	-0.168
40	0.4131	0.502	-0.422	-0.257
30	0.2500	0.304	-0.600	-0.368
20	0.1170	0.142	-0.798	-0.486
10	0.0301	0.037	-1.00	-0.61
0	0	0	-1.187	-0.724

θ_b	C_p Calculation	C_p Experiment
90	1.215	.....
80	1.133	.....
70	1.00	.....
68	.....	1.075
60	0.800	.....
50	0.536	.....
45	.....	0.42
40	0.235	.....
33.5	.....	0.05
30	-0.068	.....
20.7	.....	-0.15
20	-0.345	.....
12.3	.....	-0.58
10	-0.574	.....
6.25	.....	-0.75
0	-0.724	.....

These results are presented in Fig. 13.

PERTURBATION THEORY

For some bodies (pointed bodies or those which are treated in terms of equivalent pointed bodies, in the transonic regime for instance) and for Mach numbers other than those specified above as being ammenable to the modified Newtonian or the modified Newtonian/shock-expansion methods ($M_\infty < 0.7$), the most useful methods for solution are those of small perturbation theory as developed by Van Dyke (Ref. 21, 22, and 23) and Spreiter (Ref. 24 and 25). Somewhat simpler linearized theories have been developed by Laitone (Ref. 26 and 27) and Adams and Sears (Ref. 28).

Several texts (Ref. 29 and 30) contain explanations of and background material on perturbation theory. Van Dyke (Ref. 21 and 22) presents more detailed developments. The pertinent developments of the theory are fully presented in the cited references; only the equations directly employed in the computation of the pressure distribution will be presented here.

The application of perturbation theory begins with the description of the flow over a body by means of a velocity potential. This approach presumes irrotational flow. This idealization, however, seems to be justified by the results in the subsonic, transonic, and low-supersonic regions obtained by Van Dyke, Spreiter, and others (see Fig. 8, 14, and 15, and Ref. 21— 30).

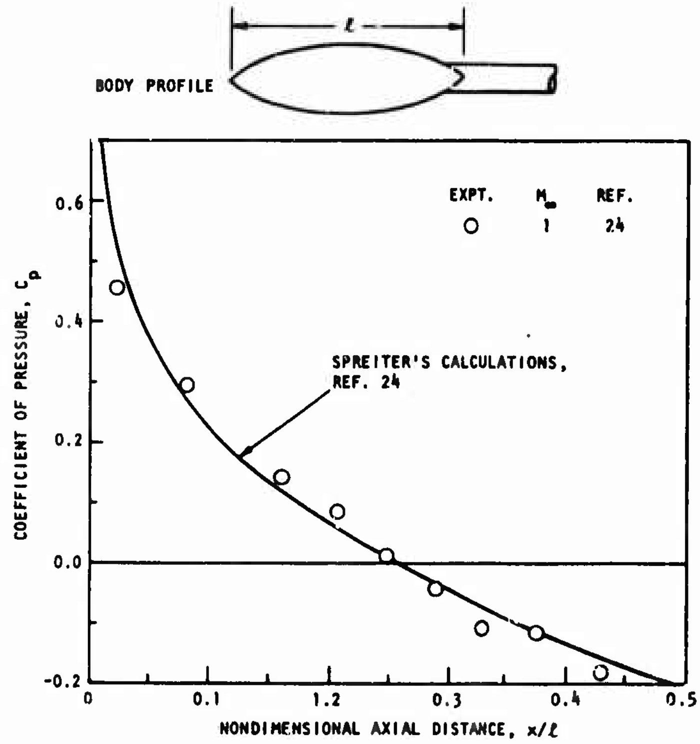


FIG. 15. Pressure Distribution on a Parabolic Arc Body, $\ell/d = 6$.

In this method, the basic flow field is described by

$$U_1 = u_\infty$$

$$U_2 = 0$$

$$U_3 = 0$$

and when the presence of a body disturbs the flow field, by

$$U_1 = u_\infty + u_1$$

$$U_2 = u_2$$

$$U_3 = u_3$$

where u_1, u_2, and u_3 are the perturbation velocities. The problem is then stated in terms of the nondimensionalized parameters,

$$u_1/u_\infty, \quad u_2/u_\infty, \quad u_3/u_\infty$$

The criterion for application of this method is that these nondimensional parameters are much less than unity. For the majority of the bodies of interest, this criteria will be met except in the region of the stagnation point of blunt bodies. Van Dyke (Ref. 21 and 23) presents a method of correction for subsonic flow that renders the solution valid even in the stagnation region if the nose contour is no more blunt than round. A rather obvious example of a body that would be excluded from treatment by perturbation methods is a flat-faced cylinder. Such a body is excluded because its nose is more blunt than round, and the ratios u_2/u_∞ and u_3/u_∞ would not be small.

The first step in the development of perturbation methods is the description of the flow field. Both velocity potential and stream functions have been used for this purpose. A perturbation velocity potential, ϕ, is most often used and defined such that

$$\frac{\partial\phi}{\partial x} = \phi_x = \frac{u_1}{u_\infty}$$

$$\frac{\partial\phi}{\partial y} = \phi_y = \frac{u_2}{u_\infty}$$

$$\frac{\partial\phi}{\partial z} = \phi_z = \frac{u_3}{u_\infty}$$

The equation of motion of the flow field about the body may then be stated in terms of this perturbation velocity potential:

$$\begin{aligned}(1 - M_\infty^2)\,\phi_{xx} + \phi_{yy} + \phi_{zz} \qquad (16)\\ = M_\infty^2\Big[\frac{\gamma-1}{2}\left(2\phi_x + \phi_x^2 + \phi_y^2 + \phi_z^2\right)\left(\phi_{xx} + \phi_{yy} + \phi_{zz}\right)\\ + 2\phi_x\phi_{xx} + \phi_x^2\phi_{xx} + \phi_y^2\phi_{yy} + \phi_z^2\phi_{zz} + 2\phi_z\left(1 + \phi_x\right)\phi_{zx}\\ + 2\phi_y\left(1 + \phi_x\right)\phi_{xy}\Big]\end{aligned}$$

Equation 16 will yield velocity components which are exact, as long as the limitation that ϕ_x, ϕ_y, ϕ_z << 1 is met. These equations, however, are quite unmanageable. Converting to a cylindrical coordinate system, reducing the equations to second order in terms of the derivatives of ϕ, and limiting consideration to bodies of revolution yields the following form:

$$\begin{aligned}\phi_{rr} + \frac{\phi_r}{r} \pm \beta^2\phi_{xx} = M_\infty^2\Bigg\{2\left[\frac{(\gamma+1)M_\infty^2}{2\beta^2} \pm 1\right]\beta^2\phi_x\phi_{xx} \qquad (17)\\ + 2\phi_r\phi_{xr} + \phi^2{}_r\phi_{rr}\Bigg\} \qquad \begin{pmatrix}+ \text{ for subsonic}\\ - \text{ for supersonic}\end{pmatrix}\end{aligned}$$

where

$$\beta = \begin{cases} \sqrt{1 - M_\infty^2} & \text{subsonic flow} \\ \sqrt{M_\infty^2 - 1} & \text{supersonic flow} \end{cases}$$

A linearized form for determining the potential, ϕ, is obtained by further approximation, neglecting the terms on the right-hand side of Eq. 17 which contains products of the derivatives of ϕ. This then results in the linear equation

$$\phi_{rr} + \frac{\phi_r}{r} \pm \beta^2\phi_{xx} = 0 \qquad (18)$$

This linearized form has been used by Laitone (Ref. 26 and 27) and by Adams and Sears (Ref. 28) in both the subsonic and supersonic flight regimes. Initially the coefficient of pressure resulting from linearized theory was expressed

$$C_p = -2\phi_x \qquad (19a)$$

However, as pointed out in Ref, 28, ϕ_x and ϕ_r are actually of different order at the body surface, and the accuracy of the method in predicting the coefficient of pressure is improved by retaining the second-order term in ϕ_r,

$$C_p = -2\phi_x - \phi_r^2 \qquad (19b)$$

This improvement comes at no increase in complexity of the solution, for at the body surface

$$\phi_r = dR/dx \qquad (20)$$

where

R = R(x) describes the variation of the cross-sectional radius, R, with axial distance, x.

It should be noted that Matthews in Ref. 31 employed Eq. 19a rather than Eq. 19b for linearized theory. For this reason his statement in that reference as to the inaccuracy of linearized theory is not correct as is pointed out by Adams and Sears in Ref. 28. Adams and Sears (Ref. 28) presented the following approximate expression for the linearized potential in the subsonic flow regime:

$$\phi(x,r) \approx \frac{1}{2\pi}\left[S'(x)\ln\left(\frac{r\beta}{2}\right) - \frac{1}{2}S'(0)\ln x - \frac{1}{2}S'(\ell)\ln(\ell-x)\right.$$
$$\left. - \frac{1}{2}\int_0^x S''(\xi)\ln(x-\xi)d\xi + \frac{1}{2}\int_x^\ell S''(\xi)\ln(\xi-x)d\xi\right] \tag{21a}$$

where

$S = \pi R^2$

$R = r(x)$

ℓ = total length of body

Notice that this solution yields singularity at the stagnation point; for this reason, the solution obtained is not valid in the region of a stagnation point. However, for points a distance $x > 0.05\ell$ away from a stagnation point, this solution gives good results. For pointed bodies, the solution is valid except very near the tip. Additionally, because Eq. 21a was developed for subsonic flow, this solution presumes that the body curvature is smooth and continuous from nose to tail. This would eliminate from consideration those configurations with cylindrical afterbodies or flat sterns. However, comparison of the results of Eq. 21a with experimental data for configurations with cylindrical afterbodies shows that the solution is good except in the region of discontinuity (Ref. 24 and 25). The corresponding supersonic flow solutions developed from linearized theory are limited to pointed bodies but the shape of the aft section has no influence. As a general rule, approximate equations such as Eq. 21a, developed from the linearized equation, Eq. 18, will obtain good results for $M_\infty < 0.7$. As an example, the pressure distribution over a prolate spheroid (an ellipsoid of revolution) in subsonic flow is calculated in the Example of Linearized Perturbation Theory, page 37. Equation 21a yields the coefficient of pressure at the body surface:

$$C_p \approx \frac{4\tau^2}{\ell^2}\left[2\ln\left(\frac{\tau\beta}{\ell}\right) + 1 + \frac{\tau^2}{R^2}\right] \tag{21b}$$

where

$$R = 2\tau/\ell \ (\ell x - x^2)^{\frac{1}{2}}$$

ℓ = total length of body

$\tau = d/\ell$

The C_p at the stagnation point is again given by Eq. 13.

Solutions obtained from the linearized equation, Eq. 18, cannot be used in the transonic regime because the term $\beta^2\phi_{xx}$ retained is no longer large in comparison to one of the neglected terms that appeared on the right-hand side of Eq. 17. To correct this difficulty, Van Dyke (Ref. 21) and Spreiter (Ref. 24 and 25) present more rigorous approaches to the solution for the perturbation velocity potential that offer considerably improved accuracy, particularly in the transonic flight regime. This improvement is obtained by including a nonlinear term on the right-hand side of the potential equation, Eq. 17:

$$(1 - M_\infty^2)\ \phi_{xx} + (1/r)\ \phi_r + \phi_{rr} = M_\infty^2(\gamma+1)\phi_x\phi_{xx} \qquad (22)$$

This equation, for a frictionless, irrotational flow, provides an improved first-order solution which is applicable to the subsonic, transonic, and supersonic regimes. Again, the equation for determining the coefficient of pressure at the body surface is

$$C_p = -2\phi_x - \phi_r^2$$

Van Dyke (Ref. 21) presents a solution to Eq. 22 and develops a correction method that renders the solution valid in the vicinity of the stagnation point of blunt bodies in subsonic flow, with the limitation on the solution being only that the ends be no more blunt than round. The basic equations of Van Dyke's solution may be presented as follows:

$$\phi(x,r) = (F + f)\ \ln(r) + (G + g) + \frac{M_\infty^2(\gamma+1)}{2(M_\infty^2 - 1)}\ F(F'\ \ln(r) + G') \qquad (23)$$

where the prime, ', denotes differentiation with respect to x, and

$$F(x) = RR'$$

$$f(x) = -\frac{M_\infty^2(\gamma+1)}{2(M_\infty^2 - 1)} FF'$$

$$G(x) = \begin{cases} F(x) \ln \dfrac{\beta}{2\sqrt{(x-a)(b-x)}} + \dfrac{1}{2}\displaystyle\int_a^b \dfrac{F(x) - F(\xi)}{|x - \xi|} d\xi , & M_\infty > 1 \\[2ex] F(x) \ln \dfrac{\beta}{2(x-a)} + \displaystyle\int_a^x \dfrac{F(x) - F(\xi)}{x-\xi} d\xi , & M_\infty < 1 \end{cases}$$

$$g(x) = \begin{cases} f(x) \ln \dfrac{\beta}{2\sqrt{(x-a)(b-x)}} + \dfrac{1}{2}\displaystyle\int_a^b \dfrac{f(x) - f(\xi)}{|x - \xi|} d\xi , & M_\infty < 1 \\[2ex] f(x) \ln \dfrac{\beta}{2(x-a)} + \displaystyle\int_a^x \dfrac{f(x) - f(\xi)}{x - \xi} d\xi , & M_\infty > 1 \end{cases}$$

As a correction factor for blunt ends in subsonic flow, Van Dyke then develops a correction factor to be added to Eq. 23:

$$C_R = \left[-\frac{1}{2}\frac{F^2(a)}{x-a} + \frac{1}{2}\frac{F^2(b)}{b-x}\right]\left(\frac{(\gamma-1)\, M_\infty^2}{4(1 - M_\infty^2)}\right)$$

which renders the solution valid in the region of the stagnation point.

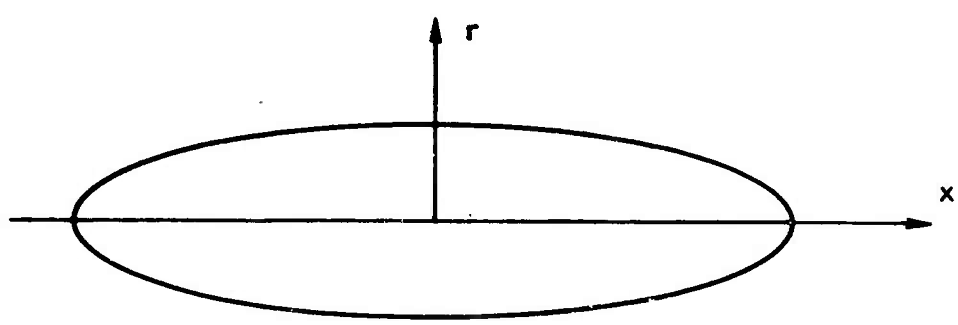
As an example of this method, Van Dyke considers the case of a prolate spheroid (Ref. 21) and arrives at the result for C_p at the body surface:

$$C_p = \tau^2 \left[2 \ln\left(\frac{\beta\tau}{2}\right) + 1 + \frac{1}{1-x^2} \right] - \left(\frac{\gamma+1}{2}\right) \frac{\tau^4}{(1 - M_\infty^{\ 2})} \left[\frac{1 - 5x^2 + 2x^4}{(1-x^2)^2} \right]$$

Adding the correction factor for the round ends gives a final expression

$$C_p = \tau^2 \left[2 \ln\left(\frac{\beta\tau}{2}\right) + 1 + \frac{1}{1-x^2} \right] - \left(\frac{\gamma+1}{4}\right) \frac{\tau^4}{(1 - M_\infty^{\ 2})} \left[\frac{1 - 11x^2 + 4x^4}{(1-x^2)^2} \right]$$

Van Dyke used the following illustrated coordinate system for his development:



$R = \sqrt{1 - x^2}$

Results obtained from Van Dyke's equation are compared with those of linearized theory in the example presented on page 37, for a prolate spheroid $\tau = 1/6$ at $M_\infty = 0.70$. This comparison indicates that for this Mach number there is no advantage in using Van Dyke's equation, for the improvement which results is very little, if any, except in the stagnation region. The use of Van Dyke's method is not so straightforward as is linearized theory, Eq. 21a. However, in the transonic regime the accuracy of Van Dyke's method improves, whereas that of linearized theory degenerates. The results from Van Dyke's second-order method are given in Fig. 14 for $M_\infty = 0.90$ and $\tau = 1/6$.

Spreiter and Alksne (Ref. 24) present a very complete development for the solution to Eq. 22 for pointed bodies of revolution. These

results are illustrated by Fig. 15, which has been reproduced from Ref. 24. Again the process of solution is rather difficult but for pointed bodies in the transonic regime, there is little alternative. For the details of this solution, the reader is referred to Ref. 24.

Van Dyke (Ref. 21) also presents a complete second-order solution. In this solution, all of the terms on the right-hand side of Eq. 17 are retained but are handled as follows:

$$\phi_{rr} + \frac{\phi_r}{r} \pm \beta^2\phi_{xx} = M_\infty{}^2\left\{2\left[\frac{(\gamma+1)\ M_\infty{}^2}{2\beta^2} \pm 1\right]\beta^2\phi_x\phi_{xx} + 2\phi_r\phi_{xr} + \phi_r{}^2\phi_{rr}\right\} \qquad \begin{pmatrix}+ \text{ for subsonic} \\ - \text{ for supersonic}\end{pmatrix}$$

where

$$\beta = \begin{cases} \sqrt{1 - M_\infty{}^2} & \text{subsonic flow} \\ \sqrt{M_\infty{}^2 - 1} & \text{supersonic flow} \end{cases}$$

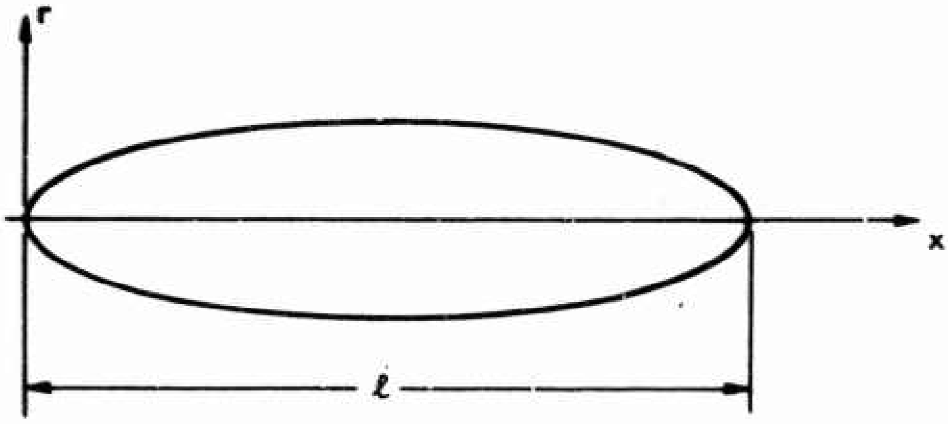
The complexities of such a solution, however, carry it beyond the scope of this report, and mention of this method is made only for the sake of completeness. Okauchi (Ref. 12) compares the results of Van Dyke's second-order solution for an ogive with data obtained from tests of the hemispherically capped ogive ($r_c/r_b \approx 0.15$), discussed previously. Representative results for low supersonic Mach numbers are included in Fig. 8.

EXAMPLE OF LINEARIZED PERTURBATION THEORY

As an example of perturbation methods, a prolate spheroid with $\tau = 1/6$ at $M_\infty = 0.70$ is considered. At the surface of a prolate spheroid, the dimensionless radius is

$$R(x) = (2\tau/\ell)\ (x\ell - x^2)^{1/2}$$

$$\tau = d/\ell$$



At the surface, $r = R(x)$; the cross-sectional area is

$$S(x) = R^2(x) = \frac{4\pi\tau^2}{\ell^2}(x\ell - x^2)$$

$$S'(x) = \frac{4\pi\tau^2}{\ell^2}(\ell - 2x)$$

$$S''(x) = -\frac{8\pi\tau^2}{\ell^2}$$

From Eq. 21a

$$\phi(x,r) \approx \left[\frac{1}{2\pi} S'(x) \ln\left(\frac{r\beta}{2}\right) - \frac{1}{2} S'(0) \ln x - \frac{1}{2} S'(\ell) \ln(\ell - x) - \frac{1}{2}\int_0^x S''(\xi) \ln(x - \xi) d\xi + \frac{1}{2}\int_x^\ell S''(\xi) \ln(\xi - x) d\xi\right]$$

for small values of r (i.e., in the vicinity of the surface). Make the substitution for S' and S":

$$\phi(x,r) \approx \frac{2\tau^2}{\ell^2}\left[(\ell - 2x)\ \ln\left(\frac{r\beta}{2}\right) - \frac{\ell}{2}\ln x + \frac{\ell}{2}\ln(\ell - x) + \int_0^x \ln(x-\xi)d\xi - \int_x^\ell \ln(\xi - x)d\xi\right]$$

Differentiate with respect to x:

$$\phi_x(x,r) \approx \frac{2\tau^2}{\ell^2}\left\{-2\ \ln\left(\frac{r\beta}{2}\right) - \frac{\ell}{2}\left(\frac{1}{x}\right) - \frac{\ell}{2}\left(\frac{1}{\ell-x}\right) + \left(\int_0^x \frac{\partial[\ln(x-\xi)]}{\partial x}\,d\xi + \ln(0) - 0\right) - \left(\int_x^\ell \frac{\partial[\ln(\xi-x)]}{\partial x}\,d\xi + 0 - \ln(0)\right)\right\}$$

$$\phi_x \approx \frac{2\tau^2}{\ell^2}\left[-2\ \ln\left(\frac{r\beta}{2}\right) - \frac{\ell^2}{2(\ell x - x^2)} + \int_0^x \frac{1}{x-\xi}\,d\xi + \int_0^\ell \frac{1}{\xi-x}\,d\xi + 2\ \ln(0)\right]$$

$$\phi_x \approx \frac{-4\tau^2}{\ell^2}\left\{\ln\left(\frac{r\beta}{2}\right) + \frac{\ell^2}{4(\ell x-x^2)} + \frac{1}{2}\{\ln(x-\xi)\}_0^x - \frac{1}{2}[\ln(\xi-x)]_x^\ell - \ln(0)\right\}$$

$$\phi_x \approx \frac{-4\tau^2}{\ell^2}\left[\ln\left(\frac{r\beta}{2}\right) + \frac{\ell^2}{4(\ell x - x^2)} - \frac{1}{2}\ln(\ell x - x^2)\right]$$

Now evaluate ϕ_x at the body surface $r = R(x) = (2\tau/\ell)(\ell x - x^2)^{\frac{1}{2}}$:

$$\phi_x(x,R) \approx \frac{-4\tau^2}{\ell^2}\left\{\ln\left(\frac{R\beta}{2}\right) + \frac{\ell^2}{4(\ell x - x^2)} - \ln\left[\frac{(2\tau/\ell)(\ell x - x^2)^{\frac{1}{2}}}{(2\tau/\ell)}\right]\right\}$$

Combine the first and last terms using the definition of R:

$$\phi_x(x,r) \approx \frac{-4\tau^2}{\ell^2}\left[\ln\left(\frac{\tau\beta}{2}\right) + \frac{\ell^2}{4(\ell x - x^2)}\right]$$

Also at the body surface,

$$\phi_r \approx \frac{\partial R}{\partial x} = \frac{(\tau/\ell)(\ell - 2x)}{(\ell x - x^2)^{\frac{1}{2}}}$$

From Eq. 19, the coefficient of pressure is

$$C_p = -2\phi_x - \phi_r^{\,2}$$

$$C_p \approx \frac{8\tau^2}{\ell^2}\left[\ln\left(\frac{\tau\beta}{\ell}\right) + \frac{\ell^2}{4(\ell x - x^2)} - \frac{(\tau^2/\ell^2)(\ell - 2x)^2}{(\ell x - x^2)}\right]$$

$$C_p \approx \frac{4\tau^2}{\ell^2}\left\{2\ln\left(\frac{\tau\beta}{\ell}\right) + \frac{2\ell^2}{4(\ell x - x^2)} - \frac{1}{4}\left[\frac{\ell^2 - 4x\ell + 4x^2}{\ell x - x^2}\right]\right\}$$

$$C_p \approx \frac{4\tau^2}{\ell^2}\left[2\ln\left(\frac{\tau\beta}{\ell}\right) + 1 + \frac{\ell^2}{4(\ell x - x^2)}\right]$$

The last term of this equation in terms of the body cross-sectional radius is expressed

$$C_p \approx \frac{4\tau^2}{\ell^2}\left[2\ \ln\left(\frac{\tau\beta}{\ell}\right) + 1 + \frac{\tau^2}{R^2}\right]$$

This may be compared with Van Dyke's solution (page 42) by substitution of $\ell = 2$ and $R = \tau\sqrt{1 - x^2}$ to fit Van Dyke's choice of coordinates. Thus the linear theory gives

$$C_p \approx \tau^2\left[2\ \ln\frac{\tau\beta}{2} + 1 + \frac{1}{1 - x^2}\right]$$

as compared to Van Dyke's solution

$$C_p = \tau^2\left[2\ \ln\frac{\tau\beta}{2} + 1 + \frac{1}{1 - x^2}\right]$$

$$- \left[\frac{\gamma+1}{4}\ \frac{\tau^4}{1 - M_\infty^2}\ \frac{1 - 11x^2 + 4x^4}{(1 - x^2)^2}\right]$$

The difference between the two equations is in the second bracket. This arises from Van Dyke's use of Eq. 22 and the correction term C_R instead of Eq. 18, which is the starting point for the development of Eq. 21a.

As a numerical example, consider a prolate spheroid with $\tau = 1/6$ at $M_\infty = 0.7$ (data from Ref. 31).

ℓ, %	C_p Linear	C_p Experimental	C_p Van Dyke
5.0	+0.0176	+0.025	+0.030
7.5	-0.029	-0.040	-0.023
12.5	-0.0654	-0.066	-0.064
25.0	-0.092	-0.096	-0.091
37.5	-0.099	-0.100	-0.101
50.0	-0.101	-0.101	-0.102

From these calculations, one sees that the more complicated methods of Van Dyke offer little advantage at $M_\infty = 0.70$. However, it should be noted that at Mach numbers in the transonic regime, the accuracy of linearized solution degenerates, whereas, the results obtained by Van Dyke's method improve in accuracy somewhat (see Fig. 14).

SIMILARITY RULES

The final method to be discussed is that of similarity. In this method, a known pressure distribution on one body is used to predict the pressure distribution on another affinely related body. Affine implies that all coordinates in a given direction change only by a uniform ratio from one member to another. For example, on bodies of revolution this affine relationship may be expressed

$$r = \tau R(x)$$

where r is the radial distance to the surface from the axis of symmetry, and τ is a characteristic body parameter such as fineness ratio or slope.

This approach may be used in two ways:

1. In subsonic, transonic, supersonic, or hypersonic flow, one may use experimental measurements on one member of an affinely related family of bodies to predict the distribution on the other members of the family.

2. In subsonic flow, one may correct exact incompressible potential flow solutions for the effects of compressibility.

Similarity rules are discussed and compared with data in Ref. 20, 29, 30, 31, and 32.

For irrotational flows, a velocity potential exists such that

$$\Phi = \Phi(x,r;\ M,\gamma,\tau)$$

The solution (Ref. 33) proceeds by separating the dependent variable into components and then stretching each component and the independent variables by factors that depend on the original parameters:

$$\Phi/u_\infty = x + \emptyset + \phi$$

where ϕ is the linearized potential theory and $\emptyset$ the second-order increment. For the linearized problem

$$(1 - M_\infty{}^2)\phi_{xx} + \phi_{rr} + \frac{\phi r}{r} = 0 \tag{25}$$

$$\phi \to 0 \text{ at infinity}$$

$$\phi_r \to \tau R'(x) \text{ at } r = \tau r(x)$$

Thus, the linearized problem is not dependent on γ and depends only on M_∞ and τ. The similarity rules are obtained by considering arbitrary scale transformation in ϕ and r. The only choice that reduces the number of parameters in the problem is

$$\phi(x, R) = 1/\beta^2 \; F(x, \rho) \tag{26}$$

$$\rho = \beta r$$

$$\beta = \begin{cases} \sqrt{1 - M_\infty{}^2} & \text{for subsonic} \\ \sqrt{M_\infty{}^2 - 1} & \text{for supersonic} \end{cases}$$

The problem then becomes

$$F_{\rho\rho} + F_\rho \phi \pm F_{xx} = 0 \tag{27}$$

$\begin{pmatrix} + \text{ for subsonic} \\ - \text{ for supersonic} \end{pmatrix}$

Thus, the original problem is reduced to one where dependence is on M_∞ and τ, not separately, but only in the combination of $\beta\tau$. The resultant similarity rule is then

$$\phi(x,r;\, M_\infty,\tau) = (1/\beta^2)\; F(x,\beta r;\beta\tau) \tag{28}$$

For the velocity potential and the coefficient of pressure to its first order, the results are

$$(1/u_\infty)\Phi(x,r;\ M_\infty,\tau) = x + (1/\beta^2)F(x,\beta r;\beta\tau) \tag{29}$$

$$C_p(x,r;\ M,\tau) = (1/\beta^2)P(x,\beta r;\beta\tau) \tag{30}$$

Two further equations used for determining the C_p in compressible flow from the incompressible $C_{p_{inc}}$ are stated by Sears (Ref. 20):

$$C_p(x,R) = C_{p_{inc}}(x,R) - (R^2)'' \ln\beta \tag{31}$$

and by Lees (Ref. 34):

$$C_p = C_{p_{inc}}\left[1 + \frac{\ln\beta}{0.31(\ln\tau)}\right]$$

Sears (Ref. 20) indicates that for slender bodies of revolution in low subsonic flow, $M_\infty < 0.7$, correction formulas such as these closely approximate theoretical results. This conclusion was reached by consideration of the results from Ref. 31 for ellipsoids of revolution with ℓ/d ratios of 6 and 10.

In the transonic flow regime, the similarity rule takes a somewhat different form. Additionally, for bodies of revolution in the neighborhood of the body surface, a similarity rule exists only if the body is very slender. In terms of determining the pressure distribution, this severe limitation makes the use of transonic similarity rules of little value (Ref. 32). However, if the bodies of interest are sufficiently slender, affinely related bodies will be similar if the similarity parameter,

$$J = \tau^2 \ln\left[\sqrt{2}\,\tau(1 - M_\infty)\right] \tag{32}$$

is equal for both bodies.

COMPUTER TECHNIQUES

Although consideration of complex computer techniques for determining the pressure distribution was not the objective of the program, it was noted during the literature survey that several computer techniques are being used with good results for blunt bodies in supersonic flow; some of these are noted. A brief comparison of theoretical and experimental results is given in Ref. 35. Methods developed by NASA (Ref. 36) and at NWL (Ref. 37) are representative of those that yield good results. Also, Ref. 38 reports a modification of a Northrop-developed program that yields good results in the Mach 1 to 3 range.

Section 3

HEAT-TRANSFER DISTRIBUTION ON BLUNT BODIES OF REVOLUTION

The analysis of the convective heating of a high-speed vehicle is an integral part of the evaluation of the aerodynamic and thermal behavior of the vehicle. Methods of analysis for convective heating have been extensively developed over the past two decades. However, this work has been directed primarily toward an understanding of heating conditions where the stagnation enthalpy is much larger than the enthalpy of the air at the surface of the vehicle.

The experimental data reviewed were those obtained for Mach numbers less than 5 both in free flight and in wind tunnel tests. Not all of the available data are reviewed since some wind tunnel conditions were not compatible with actual flight conditions expected at the corresponding Mach numbers, and some data were originally presented in such a fashion that it made comparison difficult to achieve.

Reasonable agreement between theory and experiment appears to have been achieved for flight regimes above a Mach number of 1.6 for blunt body vehicles. However, adequate experimental verification of the analytical methods developed for the high Mach number regimes has not been obtained for the range of flight Mach numbers from 0.7 to 1.6.

Since the Reynolds number encountered in the low altitude (sea level to 50,000 feet), the high-speed flight regime under consideration is expected to be high; a large portion of the boundary layer is expected to be turbulent. It will be necessary, therefore, to examine methods of analysis for the laminar, transitional, and turbulent heat-transfer regimes and to compare these methods with the appropriate experimental results.

STAGNATION-POINT HEAT TRANSFER

Various methods of computing the stagnation-point heat transfer to blunt body supersonic vehicles have been presented in the literature. Probably the best known of these methods are those of Lees (Ref. 39), Fay and Riddell (Ref. 40), Detra, Kemp, and Riddell (Ref. 41), Vaglio-Laurin (Ref. 42), and Cohen (Ref. 43). These methods were developed to predict the heat transfer to blunt-body vehicles as vehicle speeds were pushed to higher and higher values. A major objective of these studies has been to account for the thermochemistry effects that arise as a result of the higher stagnation temperatures encountered.

The equation derived by Sibulkin (Ref. 44) for the heat transfer at the stagnation point of a sphere has been used by many investigators to evaluate the heat flux at the stagnation point. Starting with Sibulkin's equation

$$Nu_L = 0.763L \ (B)^{1/2} \left(\rho_e/\mu_e\right)^{1/2} Pr^{0.4} \qquad (33)$$

where Nu_L is the Nusselt number based on a typical length, L, and B is the velocity gradient at the stagnation point, the heat transfer at the stagnation point may be written

$$\dot{q}_{w_s} = 0.763 \ (B)^{1/2} \left(\rho_e/\mu_e\right)^{1/2} \left(Pr\right)^{0.4} k\left(T_{e_s} - T_w\right) \qquad (34)$$

This expression may be modified to read

$$\dot{q}_{w_s} = 0.763 \left(Pr^{-0.6}\right) \left(\rho_{e_s} \mu_{e_s}\right)^{1/2} \left(h_{e_s} - h_w\right) (B)^{1/2} \qquad (35)$$

Substituting in the expression for the velocity gradient at the stagnation point,

$$B = \left(\frac{du_e}{ds}\right)_s \qquad (36)$$

one obtains the result

$$\dot{q}_{w_s} = 0.763 \left(Pr^{-0.6}\right) \left(\rho_{e_s} \mu_{e_s}\right)^{1/2} \left(h_{e_s} - h_w\right) \left(\frac{du_e}{ds}\right)_s^{1/2} \qquad (37)$$

where Pr = 0.71 for computations. In this expression the density, ρ_{e_s}, and the viscosity, μ_{e_s}, are evaluated at the outer edge of the boundary layer at the stagnation point of the body. Note that the only effect of the wall temperature in this equation occurs in the evaluation of the wall enthalpy.

To overcome this deficiency, Fay and Riddell (Ref. 40) obtained a correlation of numerical results with the following expression for the heat transfer at the stagnation point of a blunt body

$$\dot{q}_{w_s} = 0.763 \left(Pr^{-0.6}\right) \left(\rho_{w_s} \mu_{w_s}\right)^{0.1} \left(\rho_{e_s} \mu_{e_s}\right)^{0.4} \left(h_{a_w} - h_w\right) \left(\frac{du_e}{ds}\right)_s^{1/2} \qquad (38)$$

The velocity gradient at the stagnation point may be evaluated from the Newtonian pressure distribution:

$$\left(\frac{du_e}{ds}\right)_s = \frac{1}{R_o} \sqrt{2\left(P_{s_e} - P_\infty\right)/\rho_{s_e}} \qquad (39)$$

Hence, using the perfect gas equation, a heat-transfer parameter may be written as

$$\dot{q}_{w_s} \sqrt{R_o/P_{s_e}} = 0.763 \left(Pr^{-0.6}\right) \left(\rho_{w_s} \mu_{w_s}\right)^{0.1} \left(\rho_{e_s} \mu_{e_s}\right)^{0.4} \left(h_{a_w} - h_w\right) \qquad (40)$$

$$\left[\left(2\bar{R}T_{s_e}\right)\left(1 - \frac{P_\infty}{P_{s_e}}\right)\right]^{1/4} \frac{1}{\sqrt{P_{s_e}}}$$

Cohen (Ref. 43) has also presented a correlation of the results of numerical solutions for high-velocity heat transfer (U_∞ > 10,000 fps) in the form

$$\dot{q}_{w_s} \sqrt{R_o/P_{s_e}} = 0.767 \left(Pr_{w_s}\right)^{-0.6} \left(\rho_{w_s} \mu_{w_s}\right)^{0.07} \left(\rho_{e_s} \mu_{e_s}\right)^{0.43} \left(h_{s_e} - h_w\right) \qquad (41)$$

$$\left[\left(2\bar{R}T_{s_e}\right)\left(1 - \frac{P_\infty}{P_{s_e}}\right)\right]^{1/4} \frac{1}{\sqrt{P_{s_e}}}$$

In addition to these expressions, Detra, Kemp, and Riddell (Ref. 41) have presented a correlation for high velocity (U_∞ > 10,000 fps) in the simplified form of

$$\dot{q}_{w_s} \sqrt{R_o/P_{s_e}} = 2.28 \times 10^{-2} \left(h_{s_e}\right)^{1.075} \tag{42}$$

The experimental blunt-body heat-transfer data available for comparison with the preceeding methods of calculation are limited for flight Mach numbers between 2 and 5, whereas appropriate data below a Mach number of 1.6 are simply not available (Ref. 45, 46, 47, 48, and 49). This is shown in Table 2, which presents a summary of the appropriate blunt-body heat-transfer distribution data available. The Mach number and Reynolds number at which the data were obtained are indicated in the table. Appropriate stagnation-point heat-transfer data were obtained from the results presented by Buglia (Ref. 50), Chauvin (Ref. 51), and Chauvin and Maloney (Ref. 52). Reference should be made to the Annotated Bibliography, page 87 in the Appendix, for an indication of the difficulties encountered in using data presented in the other reports listed.

TABLE 2. Low Supersonic Blunt-Body Heat-Transfer Distribution Data

Report	Type of Test	Configuration	Mach No.	Average Surface Finish μ inch	Last Measurement Station: Distance from Stag. Point (in.)	Last Measurement Station: Local Reynolds Number	Type of Data Available: Laminar	Type of Data Available: Turbulent	Type of Data Available: Transition
Langley Research Center J. J. Buglia NASA TN D-955	Free Flight	Hemispherically Capped Cone R_N/R_B = 0.74 Cone half-angle = 14.5° R_N = 6.498"	2.32 2.47 2.63 2.80 2.97 3.14	2-5	18.20	7.42×10^6 7.66×10^6 7.84×10^6 8.36×10^6 8.54×10^6	X X X X X	X X X X X	X X X X X
Naval Ordnance Laboratory I. Korobkin Navord Report 2865	Wind Tunnel	Hemispherical D = 1 in.	2.80	Not reported	0.566	Not reported	X	X	X
Naval Ordnance Laboratory I. Korobkin Navord Report 3841	Wind Tunnel	Hemisphere cylinder D = 2 in.	1.90 2.80 3.26 4.20 4.87	Not reported	0.866	2.6×10^5 1.4×10^5 4.0×10^5 4.1×10^5 2.4×10^5	X X X X X		
Ames Research Center A. Stine and K. Wanlass, NACA TN-3344	Wind Tunnel	Hemisphere Cylinder D = 4 in.	1.97	20	4.20	2.4×10^6	X	X	
Langley Research Center L. T. Chauvin and J.P. Maloney, NACA RM-L53L 08a	Wind Tunnel	Hemisphere Cylinder D = 4 in.	1.62 2.05 2.54 3.04	"Polished"	3.414	2.2×10^6 2.6×10^6 2.1×10^6 7.0×10^5	X X X X	X X X	X X X
Lewis Research Center N. S. Diaconis, R. J. Wisniewski, and J. R. Jack NACA TN-4099	Wind Tunnel	Hemispherically Capped Cone Included angle= 90°30' D = 1.4 in.	3.12 3.12	16 130	2.94	R_{e_∞} = 1.04 x 10	X X		X X
Langley Research Center, I. E. Beckwith, J. J. Gallagher NACA TN-4125	Wind Tunnel	Sphere D = 3½ in. Hemisphere-cylinder D = 2 in.	2.00	Not reported	3.5 1.57	R_{e_∞} = 4.45 x 10 R_{e_∞} = 0.85 x 10	X X	X X	X X

It would appear, at this point, that the results of the experimental program reported by Buglia (Ref. 50) represent the most complete set of experimental data obtained for free-flight or wind tunnel environments. For this reason, the set of results obtained by Buglia has been employed for comparison with the formulas previously presented.

In the flight test conducted by Buglia, heat-transfer and transition data on a highly polished hemisphere cone were measured over a Mach number range from 2.32 to 3.14 up to a free-stream Reynolds number of 24 x 10^6 based on the body diameter. The hemisphere radius was 6.498 inches and the cone half-angle was 14.5 degrees. The stagnation-point heat-transfer results obtained from this experiment are shown in Fig. 16, plotted as the heat-transfer parameter, $\dot{q}_{w_s} \sqrt{R_o/P_{s_e}}$, versus the flight velocity, U_∞. Here $\dot{q}_{w_s}$ is the heating rate, R_o is the nose radius, and P_{s_e} is the stagnation pressure at the nose of the vehicle. Also shown in Fig. 16 are experimental data obtained in a shock tube, as reported by Rose and Stark (Ref. 53).

The theoretical prediction of Fay and Riddell, obtained from Eq. 40, is shown in Fig. 16. For comparison the correlation presented by Detra, Kemp, and Riddell is also shown in Fig. 16.

It is apparent from Fig. 16 that the expression developed by Fay and Riddell can be used with confidence down to velocities of the order of 2,000 fps if the wall enthalpy is properly considered.

Table 3 presents tabulated values of the experimental results obtained by Buglia (Ref. 50) for the stagnation-point heat-transfer rates for the flight Mach numbers of 2.32, 2.47, 2.63, 2.80, 2.97, and 3.14. Also presented are values of the stagnation-point heat-transfer rates calculated from first, Fay and Riddell according to Eq. 40; second, from Detra, Kemp, and Riddell, according to Eq. 42; and finally, from Cohen, according to Eq. 41.

For this limited range of results, the Fay and Riddell equation indicates an average error of 13% high, whereas, Eq. 41 from Cohen indicates an average error of 10%.

For conditions where the enthalpy at the wall temperature becomes of the same order as the stagnation enthalpy, experimental results presented by Chauvin (Ref. 51) and Chauvin and Maloney (Ref. 52) are of interest. The heat transfer to a 4-inch hemisphere was measured for several flow Mach numbers and stagnation temperatures. The stagnation-point heat-transfer rates are presented in Fig. 17 along with the data of Buglia (Ref. 50) as functions of the enthalpy drop across the boundary layer at the stagnation point. These results may be compared with the theoretical results of Fay and Riddell for a Lewis number of unity, as shown in Fig. 17.

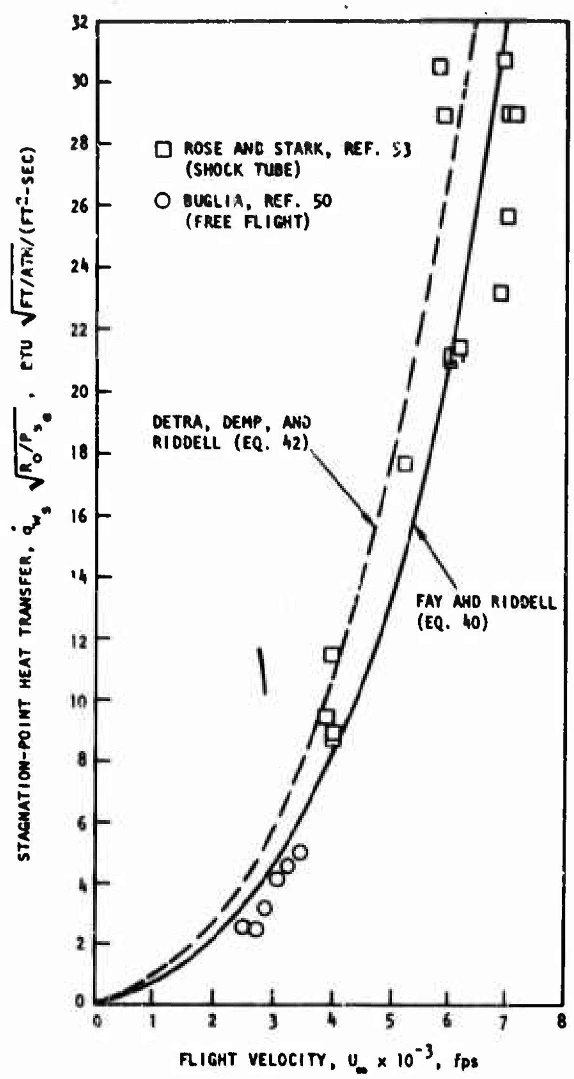


FIG. 16. Stagnation-Point Heat Transfer for Air.

It thus becomes apparent that additional blunt-body stagnation-point heat-transfer data are required for flight Mach numbers in the range of 0.7 to 2.0 in order to verify the applicability of the analytical methods.

LAMINAR HEAT TRANSFER OVER SPHERICAL SECTIONS

The distribution of heat transfer over the surface of the hemisphere in the region of the stagnation point and along the surface of the cone section of the vehicle examined by Buglia can be compared in the following way.

TABLE 3. Experimental and Theoretical Stagnation-Point Heat Transfer for a Hemispherically Capped Cone

R_o = 6.498 inches; highly polished surface.

M_∞	U_∞, fps	T_{se}, °R	P_{se}, psia	Δh, Btu/lbm	$\dot{q}_{w_s}\sqrt{\frac{R_o}{P_{se}}}$ Btu$\sqrt{ft/atm}$/ft sec: Buglia[a]	Fay[b] and Riddell	Detra[c] and Riddell	Cohen[d]
2.32	2543	1040	81.5	77.4	2.58	2.78	3.90	2.70
2.47	2707	1110	90.0	88.2	2.46	2.65	4.83	2.32
2.63	2880	1189	99.7	97.3	3.17	3.55	5.52	3.47
2.80	3062	1278	110.5	127.3	4.04	4.66	6.34	4.59
2.97	3244	1372	121.1	142.7	4.54	5.30	7.18	5.18
3.14	3425	1471	132.3	160.5	4.96	5.99	8.03	5.88

[a]Ref. 50; [b]Ref. 40; [c]Ref. 41; [d]Ref. 43.

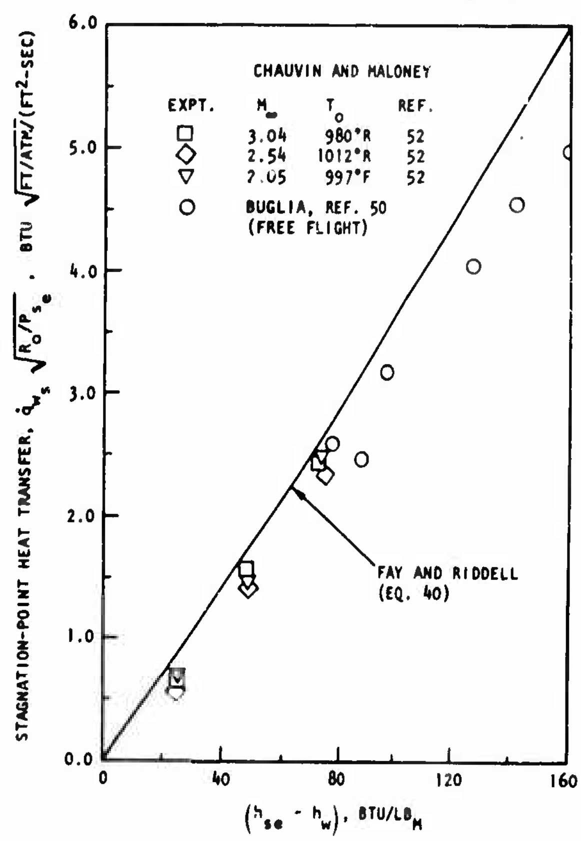


FIG. 17. Stagnation-Point Heat Transfer for Air.

Fay and Riddell, and Cohen found that the variation in fluid properties between wall conditions and stagnation conditions at the outer edge of the boundary layer must be taken into account in computing the heat-transfer distribution. One method for accounting for variable fluid properties has been advanced by Eckert (see Ref. 54 for a thorough discussion and an extensive bibliography of early work) through the introduction of the reference temperature, defined as

$$T^* = T_e + 0.50\left(T_w - T_e\right) + 0.22\left(T_{aw} - T_e\right) \tag{43}$$

where the adiabatic wall temperature is given by

$$T_{aw} = T_e\left(1 + \frac{\gamma - 1}{2} M_e^2\, r^*\right) \tag{44}$$

the recovery factor, $r^* = Pr^{1/2}$, for the laminar boundary layer, and $r^* = Pr^{1/3}$ for the turbulent boundary layer.

The laminar heat-transfer distribution over the surface of the hemisphere, as predicted by the various methods of analysis, are compared on the basis of the local Stanton number, defined as

$$C_H^* = \dot{q}_w / \left[\rho^* u_e\left(h_{aw} - h_w\right)\right] \tag{45}$$

where the reference density, ρ^*, is computed through the perfect gas equation using the static pressure at the outer edge of the boundary layer and the local reference temperature.

The methods of analysis for predicting the heat-transfer distribution over the hemisphere surface, which appear to be the most straightforward and yet within acceptable agreement with the experimental data, are the methods of Korobkin (Ref. 47) and Lees (Ref. 39).

From Sibulkin's equation for the stagnation-point heat transfer, Eq. 33, Korobkin replaced the length, L, by the wetted distance from the stagnation point, s, and the velocity gradient, β, by u_e/s and obtained the equation

$$Nu_s = 0.763\left(\frac{\rho_e u_e s}{\mu_e}\right)^{1/2} Pr^{0.4} \tag{46}$$

where the properties are evaluated at the outer edge of the boundary layer. This expression may also be written in terms of the Stanton number by dividing through by Pr Re_s where the Reynolds number, based on the conditions at the edge of the boundary layer, is written

$$Re_s = \frac{\rho_e u_e s}{\mu_e} \tag{47}$$

Korobkin thus obtained the expression

$$C_H = 0.763 \left(Pr^{-0.6}\right) \left(Re_s\right)^{-\frac{1}{2}} \tag{48}$$

for the Stanton number in the stagnation region. To account for the effect of wall temperature on the gas properties, this expression may be written in terms of properties evaluated at the reference enthalpy:

$$C_H^* = 0.763 \left(Pr^*\right)^{-0.6} \left(Re_s^*\right)^{-\frac{1}{2}} \tag{49}$$

for the stagnation region of a hemisphere. This relationship is conservative in the computation of the heat-transfer distribution over the hemisphere surface.

Lees (Ref. 39) presents the following expression for the ratio of the rate of heat transfer to a given point on the hemisphere surface to the heat transfer at the stagnation point:

$$\frac{\dot{q}_w}{\dot{q}_{w_s}} = \frac{2\theta \sin\theta \left[(1 - 1/\gamma M_\infty^2) \cos^2\theta + 1/\gamma M_\infty^2\right]}{\left[f(\theta)\right]^{\frac{1}{2}}} \tag{50}$$

where

$$f(\theta) = \left(1 - \frac{1}{\gamma M_\infty^{\;2}}\right)\left(\theta^2 - \frac{\theta \sin 4\theta}{2} + \frac{1 - \cos 4\theta}{8}\right) \tag{51}$$

$$+ \frac{4}{\gamma M_\infty^{\;2}}\left(\theta^2 - \theta \sin 2\theta + \frac{1 - \cos 2\theta}{2}\right)$$

Here θ is the angle between the flight velocity vector and the hemisphere radius vector. This expression is applicable only in a region of laminar boundary layer on the hemisphere surface.

Thus, for the stagnation region of a blunt-body vehicle, a comparison is presented of the equation of Korobkin, Eq. 49, based on reference properties, and the equation of Lees, Eq. 50, with appropriate experimental results.

For blunt-body vehicles that include either conical sections or cylindrical sections, the flat-plate formula presented by Van Driest (Ref. 55) should be available. The flat-plate heat-transfer equation for laminar flow is written

$$C_H^* = 0.332 \left(Pr^*\right)^{-2/3} \left(Re_s^*\right)^{-\frac{1}{2}} \tag{52}$$

where the properties are determined at the reference temperature and the static pressure at the outer edge of the boundary layer.

The experimental data obtained by Buglia (Ref. 50) for the heat-transfer distribution over the hemisphere and cone surfaces were reported in the form

$$C_{H_\infty} = \frac{\dot{q}_w}{\rho_\infty u_\infty \left(h_{a_w} - h_w\right)} \tag{53}$$

and were corrected to the reference temperature base by

$$C_{H_{exp}}^* = C_{H_\infty} \left(\frac{\rho_\infty u_\infty}{\rho^* u_e}\right) \tag{54}$$

The experimental values obtained by Buglia (Ref. 50) are presented in Fig. 18(a) through 18(f) as a function of the Reynolds number, defined as

$$Re_s^* = \frac{\rho^* u_e s}{\mu^*} \tag{55}$$

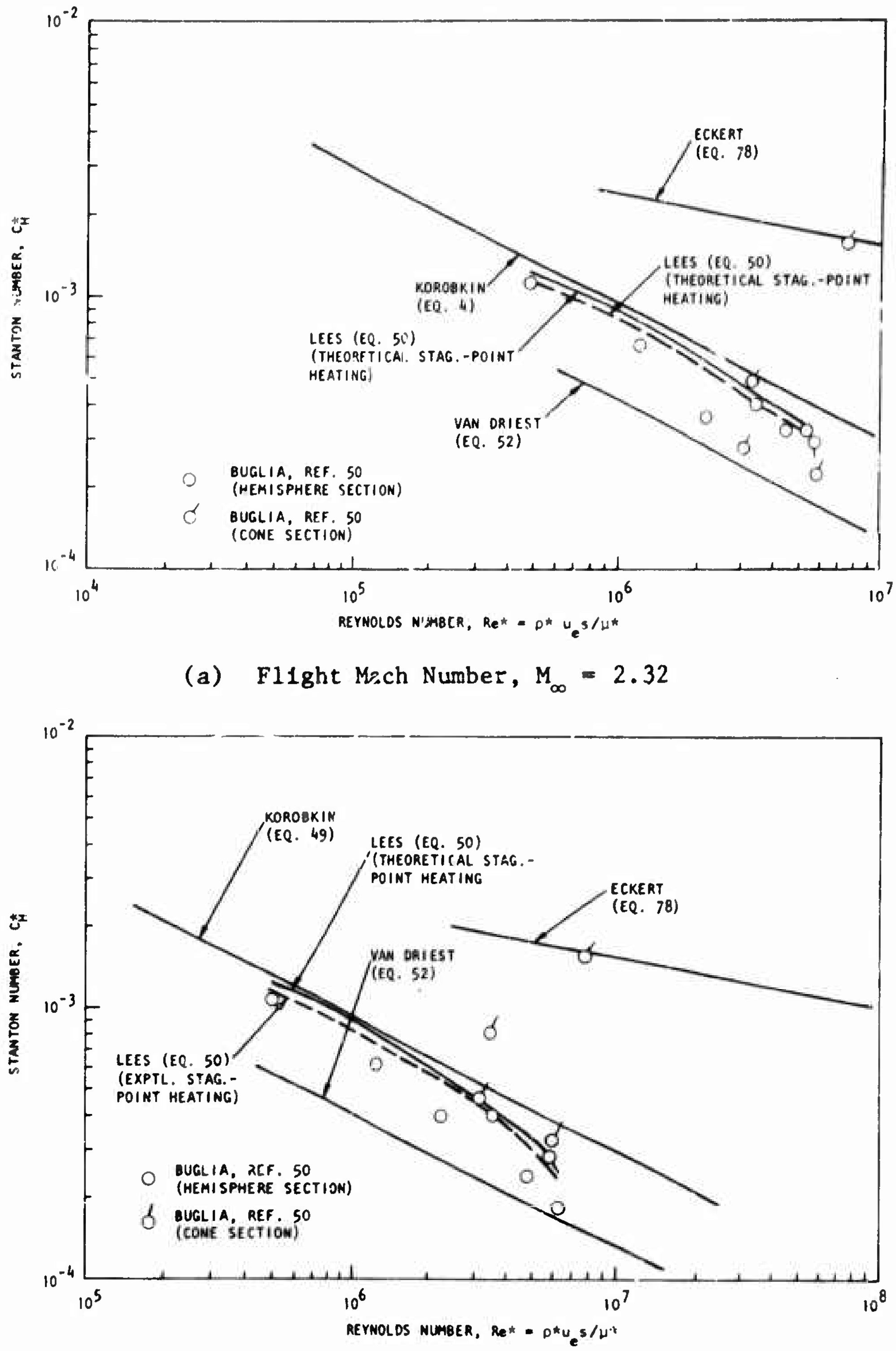
where s is the wetted length from the stagnation point to the point in question. The static pressure at the edge of the boundary layer, employed in the calculation of the local temperature and velocity, was obtained from the experimental pressure distribution. Isentropic expansion from the stagnation point to the local pressure was then assumed to calculate the local temperature and velocity. The data points presented in Fig. 18(a) through 18(f) on the cone section are indicated with a flag. It is interesting to note in the vicinity of the juncture between hemisphere and conical sections that the Reynolds number, based on reference conditions, decreases. As the flow is compressed in the region of the conical section, the boundary layer undergoes transition from laminar to turbulent flow, thus showing a significant increase in the heat-transfer rate.

The theoretical predictions of Korobkin, Eq. 49; Lees, Eq. 50; and Van Driest, Eq. 52, are also included in each of Fig. 18(a) through 18(f) for comparison with the experimental results.

The results presented in Fig. 18(a) through 18(f) indicate that the Lees relationship, based on reference values, more closely predicts the experimental data for laminar flow over both the hemisphere and the initial portion of the cone section of the body. Also, for a case where laminar flow persists over a significant portion of the conical section, it is apparent that the Van Driest equation, Eq. 52, may be used to predict the heat transfer.

It is also apparent that for the Mach number range from 2 to 3, if laminar flow persists over the entire hemisphere section, Lees' expression will account adequately for the pressure gradient effect over the juncture between the hemisphere and the conical portion of the body.

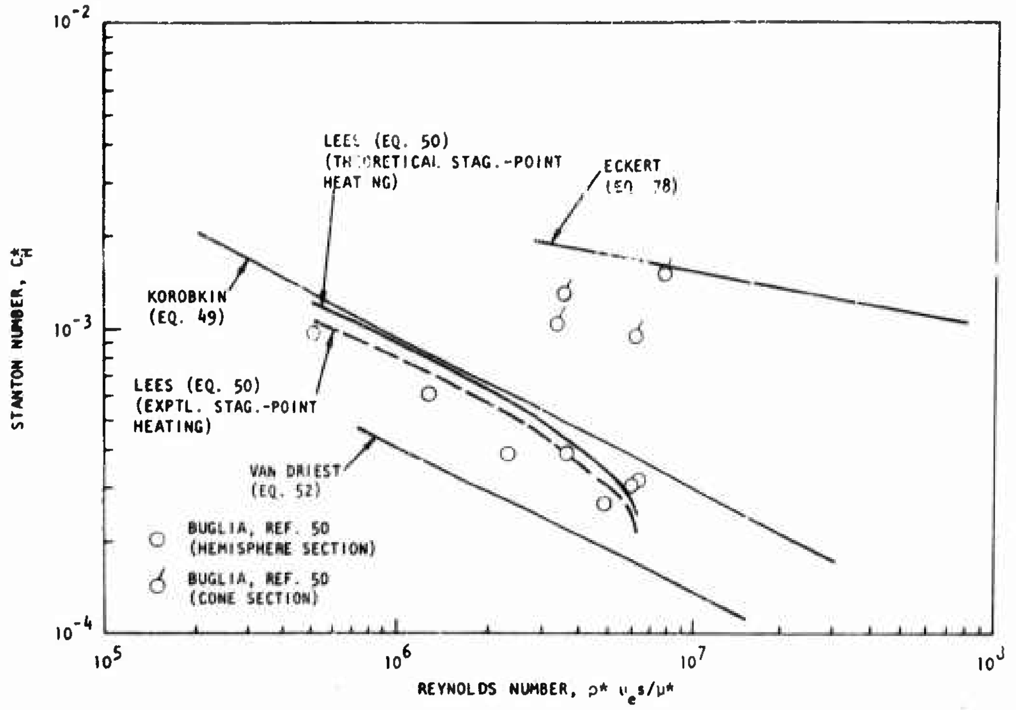
The numerical computations corresponding to the results presented in Fig. 18(a) through 18(f) are given in Tables 4(a) through 4(f). Each table contains the numerical results presented in the corresponding figure. The parameters of interest, for example, the local Reynolds number based on wetted length and reference conditions, are presented in terms of the non-dimensional wetted distance from the stagnation point for each Mach number. The results presented in these tables indicate that the best agreement with the experimental Stanton number distribution is achieved by using the Fay and Riddell stagnation-point heat-transfer equation, Eq. 40, with Lees' expression for the distribution of the local heat-transfer equation, Eq. 50, and the density of the air computed with the value of the reference temperature and local static pressure. This method will be presented in detail in a later section.



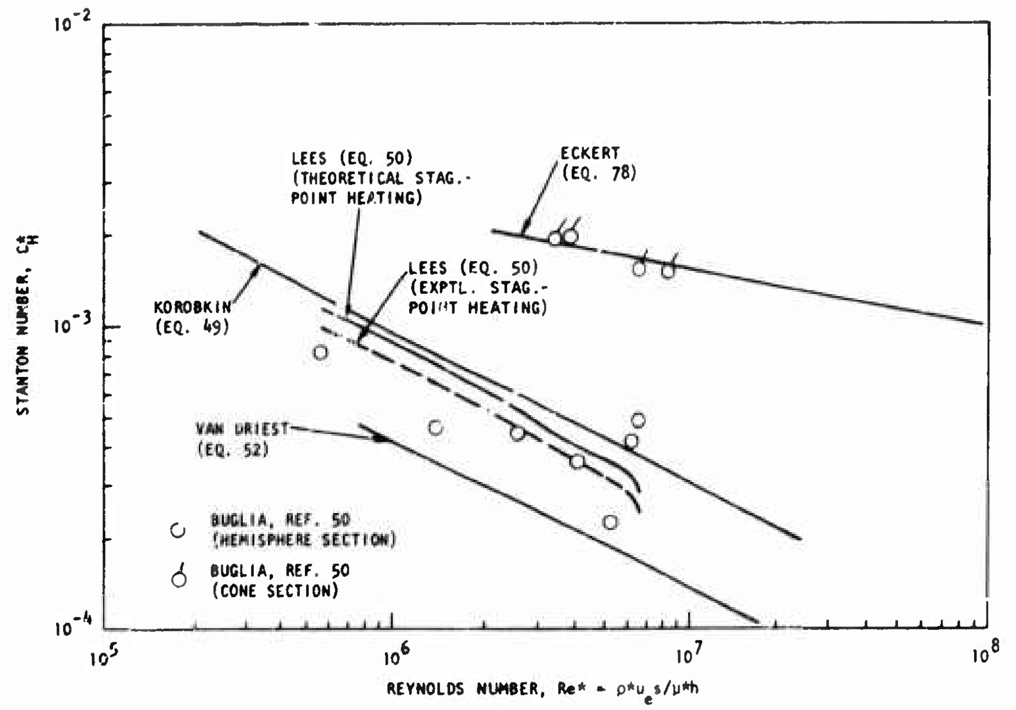
(a) Flight Mach Number, $M_\infty = 2.32$

(b) Flight Mach Number, $M_\infty = 2.47$

FIG. 18. Heat-Transfer Distribution Over a Smooth Hemispherically Capped Cone for Flight Mach Numbers; Buglia, Ref. 50.

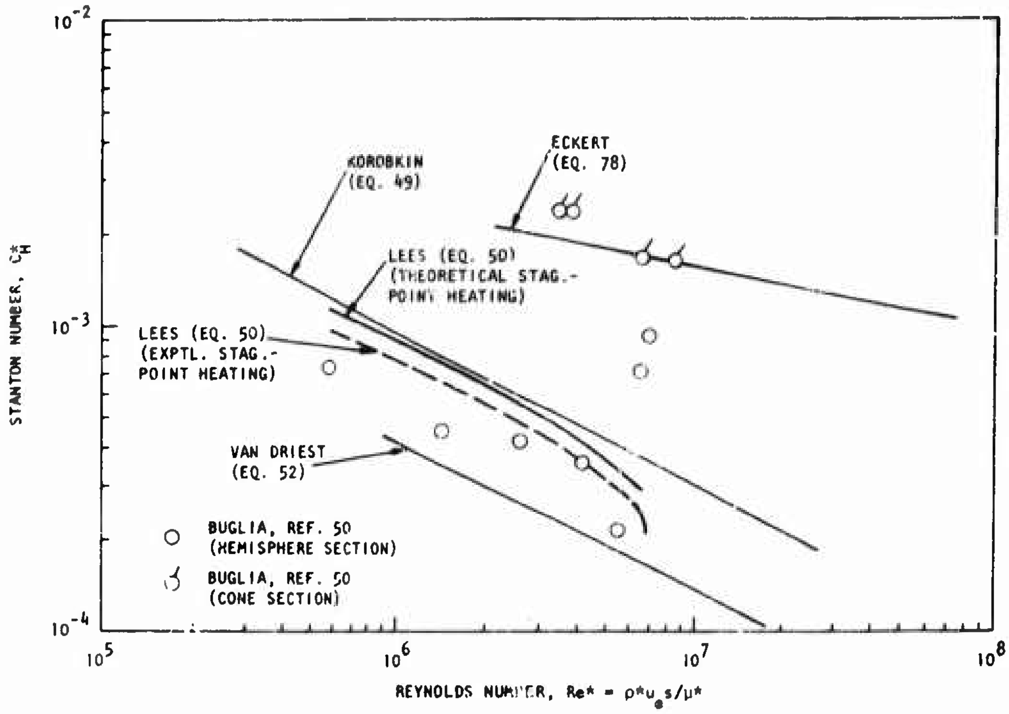


(c) Flight Mach Number, M_∞ = 2.63

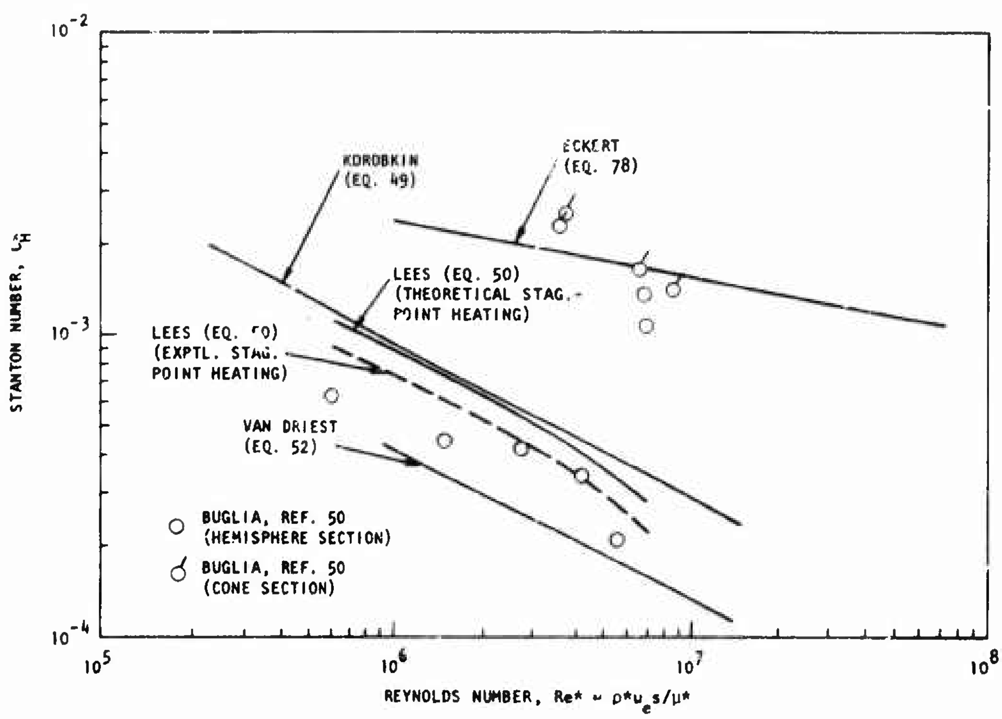


(d) Flight Mach Number, M_∞ = 2.80

FIG. 18. (Continued).



(e) Flight Mach Number, M_∞ = 2.97



(f) Flight Mach Number, M_∞ = 3.14

FIG. 18. (Continued).

TABLE 4(a). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 2.32$, $P_\infty = 11.0$ psia, $T_\infty = 500.8°R$

$s/2\ R_o$	R^*_{es} x 10^{-6}	$C^*_{H_{exp}}$ (1) x 10^3	$C^*_{H_L}$ (2) x 10^3	$C_{H_{L_e}}$ (3) x 10^3	$C^*_{H_{L_{R_E}}}$ (4) x 10^3	$C_{H_{L_{e_E}}}$ (5) x 10^3
0.090	0.479	1.15	1.242	1.458	1.155	1.355
0.150	1.195	0.67	0.805	0.942	0.748	0.876
0.210	2.162	0.37	0.592	0.686	0.550	0.636
0.270	3.411	0.40	0.440	0.502	0.409	0.466
0.330	4.482	0.32	0.377	0.419	0.351	0.389
0.390	5.325	0.32	0.335	0.358	0.312	0.333
0.520	5.696	0.29	0.275	0.267	0.256	0.249
0.660	3.056	0.28	0.329	0.264	0.306	0.246
0.810	3.304	0.50	0.332	0.259	0.309	0.241
1.050	5.758	0.22	0.186	0.153	0.173	0.142
1.400	7.415	1.57	0.215	0.170	0.200	0.158

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e\left(h_{aw} - h_w\right)}$; Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

TABLE 4(b). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 2.47$, $P_\infty = 10.8$ psia, $T_\infty = 500°R$

$s/2\ R_o$	R^*_{es} x 10^{-6}	$C^*_{H_{exp}}$ (1) x 10^3	$C^*_{H_L}$ (2) x 10^3	$C_{H_{L_e}}$ (3) x 10^3	$C^*_{H_{L_{R_E}}}$ (4) x 10^3	$C^*_{H_{L_{e_E}}}$ (5) x 10^3
0.090	0.499	1.09	1.221	1.450	1.132	1.345
0.150	1.245	0.62	0.794	0.939	0.736	0.871
0.210	2.252	0.40	0.585	0.685	0.542	0.635
0.270	3.552	0.41	0.437	0.505	0.406	0.468
0.330	4.800	0.25	0.344	0.393	0.319	0.364
0.390	5.852	0.29	0.285	0.318	0.264	0.295
0.520	6.216	0.18	0.250	0.252	0.232	0.234
0.660	3.129	0.47	0.368	0.297	0.342	0.275
0.810	3.404	0.81	0.375	0.296	0.348	0.274
1.050	5.944	0.33	0.191	0.158	0.177	0.147
1.400	7.655	1.55	0.220	0.176	0.204	0.163

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e \left(h_{aw} - h_w\right)}$; Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

TABLE 4(c). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 2.63$, $P_\infty = 10.6$ psia, $T_\infty = 499°R$

$s/2\ R_o$	R^*_{es} x 10^{-6}	$C^*_{H_{exp}}$ (1) x 10^3	$C^*_{H_L}$ (2) x 10^3	$C_{H_{L_e}}$ (3) x 10^3	$C^*_{H_{L_{RE}}}$ (4) x 10^3	$C_{H_{L_{e_E}}}$ (5) x 10^3
0.090	0.515	0.97	1.204	1.439	1.076	1.285
0.150	1.271	0.61	0.810	0.959	0.724	0.857
0.210	2.300	0.39	0.596	0.698	0.532	0.623
0.270	3.626	0.40	0.445	0.513	0.397	0.458
0.330	4.898	0.27	0.348	0.397	0.311	0.355
0.390	5.969	0.30	0.286	0.320	0.256	0.286
0.520	6.334	0.32	0.243	0.246	0.217	0.219
0.660	3.342	1.05	0.302	0.250	0.270	0.224
0.810	3.554	1.30	0.308	0.246	0.275	0.220
1.050	6.088	0.96	0.193	0.160	0.172	0.143
1.400	7.844	1.53	0.222	0.178	0.198	0.159

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e\left(h_{aw} - h_w\right)}$; Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

TABLE 4(d). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 2.80, \quad P_\infty = 10.5 \text{ psia}, \quad T_\infty = 498°R$

$s/2R_o$	$R^*_{es} \times 10^{-6}$	$C^*_{H_{exp}}{}^{(1)} \times 10^3$	$C^*_{H_L}{}^{(2)} \times 10^3$	$C_{H_{L_e}}{}^{(3)} \times 10^3$	$C^*_{H_{LR_E}}{}^{(4)} \times 10^3$	$C_{H_{Le_E}}{}^{(5)} \times 10^3$
0.090	0.563	0.82	1.150	1.434	0.997	1.244
0.150	1.404	0.46	0.752	0.934	0.652	0.810
0.210	2.565	0.43	0.540	0.667	0.468	0.578
0.270	4.038	0.36	0.408	0.496	0.354	0.430
0.330	5.287	0.22	0.352	0.416	0.305	0.360
0.390	6.191	0.42	0.324	0.366	0.281	0.317
0.520	6.496	0.48	0.280	0.284	0.243	0.246
0.660	3.412	1.94	0.336	0.280	0.292	0.243
0.810	3.703	1.95	0.325	0.264	0.281	0.229
1.050	6.500	1.52	0.210	0.180	0.182	0.156
1.400	8.362	1.50	0.239	0.197	0.207	0.171

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e \left(h_{aw} - h_w\right)}$: Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

TABLE 4(e). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 2.97, \quad P_\infty = 10.2 \text{ psia}, \quad T_\infty = 497°R$

$s/2\,R_o$	R^*_{es} $\times 10^{-6}$	$C^*_{H_{exp}}$ (1) $\times 10^3$	$C^*_{H_L}$ (2) $\times 10^3$	$C_{H_{L_e}}$ (3) $\times 10^3$	$C^*_{H_{L_{RE}}}$ (4) $\times 10^3$	$C_{H_{L_{e_E}}}$ (5) $\times 10^3$
0.090	0.581	0.74	1.134	1.432	0.972	1.227
0.150	1.433	0.45	0.764	0.955	0.655	0.818
0.210	2.589	0.43	0.564	0.698	0.484	0.598
0.270	4.117	0.36	0.414	0.507	0.355	0.435
0.330	5.497	0.22	0.339	0.408	0.291	0.350
0.390	6.552	0.71	0.298	0.346	0.255	0.296
0.520	6.958	0.94	0.249	0.262	0.214	0.225
0.660	3.428	2.36	0.332	0.277	0.285	0.237
0.810	3.720	2.36	0.314	0.256	0.270	0.219
1.050	6.585	1.69	0.217	0.187	0.186	0.160
1.400	8.535	1.62	0.243	0.202	0.208	0.173

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e\left(h_{aw} - h_w\right)}$; Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

TABLE 4(f). Experimental and Theoretical Heat-Transfer Distributions Over a Hemispherically Capped Cone

$M_\infty = 3.14$, $P_\infty = 10.0$ psia, $T_\infty = 495°R$

$s/2R_o$	R^*_{es} $\times 10^{-6}$	$C^*_{H_{exp}}$ (1) $\times 10^3$	$C^*_{H_L}$ (2) $\times 10^3$	$C_{H_{L_e}}$ (3) $\times 10^3$	$C^*_{H_{L_{R_E}}}$ (4) $\times 10^3$	$C_{H_{L_{e_E}}}$ (5) $\times 10^3$
0.090	0.598	0.63	1.118	1.429	0.925	1.183
0.150	1.491	0.45	0.732	0.933	0.606	0.772
0.210	2.693	0.42	0.542	0.682	0.448	0.565
0.270	4.194	0.35	0.421	0.518	0.348	0.429
0.330	5.597	0.21	0.344	0.417	0.285	0.345
0.390	6.867	1.04	0.281	0.335	0.233	0.277
0.520	6.766	1.344	0.278	0.287	0.230	0.257
0.660	3.419	2.29	0.334	0.277	0.276	0.229
0.810	3.683	2.52	0.313	0.252	0.259	0.209
1.050	6.476	1.68	0.236	0.201	0.196	0.166
1.400	8.584	1.40	0.254	0.211	0.210	0.175

(1) Experimental Stanton Number, $C^*_{H_{exp}} = \dfrac{\dot{q}_{w_{exp}}}{\rho^* u_e\left(h_{aw} - h_w\right)}$; Buglia, Ref. 50.

(2) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(3) Stanton Number based on the Fay and Riddell stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the edge of the boundary layer.

(4) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and reference conditions.

(5) Stanton Number based on the experimental stagnation-point heat transfer, Lees' distribution of local heat transfer, and conditions at the outer edge of the boundary layer.

It should be pointed out that the experimental results used for the comparisons made above were obtained on a smooth (roughness from 2 to 5 μ-inches) hemispherically capped cone. The polished surface condition apparently delayed transition from laminar to turbulent flow for this particular vehicle; hence, the Reynolds numbers for which laminar flow persists are high, on the order of 5 x 10^6. These data are thus presented only to compare the various methods of predicting laminar blunt-body heat-transfer distributions with available experimental results.

Additional experimental data for the heat-transfer distribution over hemispherically capped cylinders have been reported by Chauvin and Maloney (Ref. 52) and Korobkin (Ref. 47). However, an insufficient amount of supporting test information prohibited comparison of these data with the methods previously discussed. These data are, therefore, presented in Fig. 19 as the experimental Stanton number based on local conditions at the edge of the boundary layer versus the local Reynolds number. The Mach number range covered by the wind tunnel tests of Chauvin and Maloney was from 1.62 to 3.04, while Korobkin obtained a Mach number range from 1.90 to 4.87. These data indicate a fairly uniform distribution about a line predicted by Korobkin's equation, Eq. 49. It should be noted that in the tests reported by Chauvin and Maloney, and Korobkin, the boundary layer underwent transition from laminar to turbulent at a local Reynolds number of approximately 1 x 10^6, based upon the wetted length from the stagnation point and the properties evaluated at local conditions at the edge of the boundary-layer heat-transfer.

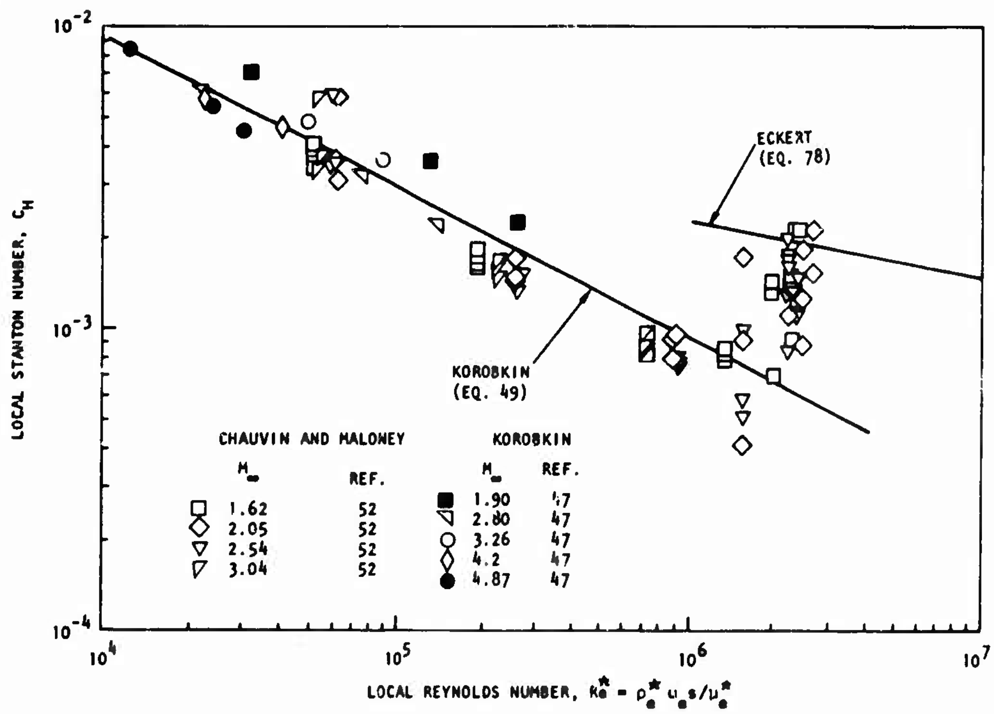


FIG. 19. Local Stanton Number Distribution Over a Hemispherically Capped Cylinder Obtained in Wind-Tunnel Tests; Chauvin and Maloney, Ref. 52, and Korobkin, Ref. 47.

Cresci, MacKenzie, and Libby (Ref. 57) have compared laminar heat-transfer data with the theory of Lees (Ref. 39) by using the stagnation-point value of Fay, Riddell, and Kemp (Ref. 56) for Mach numbers above 3. The data are presented in Ref. 57 in terms of a Nusselt number defined by

$$Nu = \dot{q}_w \left(c_p\right)_{s_e} R_o/k_{s_e}\left(h_{s_e} - h_w\right) \tag{56}$$

and a Reynolds number parameter

$$\tilde{Re} = Re\sqrt{P_s/\rho_{s_e} h_{s_e}} = Re\,\sigma_{s_e}^{1/2} \tag{57}$$

where

$$Re = \rho_{s_e}\sqrt{h_{s_e}}\,R_o/\mu_{s_e} \tag{58}$$

and

$$\sigma_{s_e} = P_s/\rho_{s_e} h_{s_e} \tag{59}$$

In terms of these parameters, the heat-transfer distribution is represented by

$$\frac{Nu}{\tilde{Re}^{1/2}} = 0.33\left(\frac{\rho_w \mu_w}{\rho_{s_e}\mu_{s_e}}\right)^{0.1}\frac{\bar{\rho}\,\bar{\mu}\,\bar{u}\,\bar{r}}{\sigma_{s_e}^{1/4}\left[\int_0^{\bar{s}}\bar{\rho}\,\bar{\mu}\,\bar{u}\,(\bar{r})^2\,d\bar{s}\right]^{1/2}} \tag{60}$$

Corresponding to this equation, the Reynolds number based on momentum thickness is given by the theory of Lees:

$$\frac{Re_\theta}{\tilde{Re}^{1/2}} = \frac{0.66\left[\int_o^{\bar{s}} \bar{\rho}\ \bar{\mu}\ \bar{u}\ (\bar{r})^2\ d\bar{s}\right]^{1/2}}{\bar{\mu}\ \bar{r}\left(\sigma_{s_e}\right)^{1/6}} \tag{61}$$

where

$$Re_\theta = \frac{\rho_e u_e \theta}{\mu_e} \tag{62}$$

and

$$\theta = \int_o^\infty \left(\rho\mu/\rho_e\mu_e\right)\left[1 - \left(u/u_e\right)\right] dy \tag{63}$$

Comparison of Eq. 55 and experimental data is presented in Ref. 57 with excellent agreement. Since these experimental results corresponded to a flight Mach number of approximately 20, they are not presented here.

HEAT TRANSFER IN THE TRANSITION REGION

An extensive discussion of boundary-layer transition is presented by Wilson in Ref. 58. This work, however, is largely concerned with transition phenomenon on pointed bodies of revolution. In fact, the observation is made in this reference that few data are available for blunt-body heat transfer in the transition region. Despite this, a computational procedure has been compared with experimental results for the heat transfer in the transition regime by Cresci, MacKenzie, and Libby (Ref. 57), by Nardo and Sadler (Ref. 59), and by Stetson (Ref. 60). In anticipation of its use in transition flow, the momentum equation for the axisymmetric boundary layer can be written in the form

$$dRe_\theta/d\bar{s} = \left(\tilde{Re}/\sigma_{s_e}^{1/2}\right)\left(\overline{\rho u}/\bar{\mu}\right)\left(C_f/2\right) - Re_\theta d(\ln \bar{r}\ \bar{\mu})/d\bar{s} \tag{64}$$

where $C_f = 2\tau_w/\rho_e u_e^2$, the local skin-friction coefficient $\sigma_{s_e} = P_s/\rho_{s_e} h_{s_e}$.
The analysis of Ref. 57 utilizes, for fully developed turbulent flows, the extension of the local Prandtl incompressible skin-friction law for the incompressible "cold-wall" case in the form

$$C_f/2 = \bar{\mu}(0.013)\left(Re_\theta\right)^{-1/4} \tag{65}$$

Numerical integration of Eq. 64 along with a skin-friction law equivalent to Eq. 65 is required. In accordance with the suggestion of Persch (Ref. 61) and including the modification suggested by Ref. 55, the skin-friction becomes

$$C_f/2 = \bar{\mu}\left[0.013\ Re_\theta^{-1/4} - \Pi\ Re_\theta^{-2}\right] \tag{66}$$

The constant Π in this expression is equivalent from the laminar flow such that the friction coefficient, $C_f/2$, remains continuous at the transition point (Ref. 57),

$$\Pi = \left[Re_\theta^{2}\left(\frac{0.013}{Re_\theta^{1/4}} - \frac{C_f}{2}\frac{\mu_{s_e}}{\mu_e}\right)\right]_{trans.} \tag{67}$$

If Eq. 64 and 66 are combined, and Eq. 64 is integrated from the transition point, then the skin-friction coefficient and the momentum-thickness distribution are known. When Reynolds analogy is modified to account for nonunity Prandtl number effects, then the heat-transfer distribution may be found from

$$\dot{q}_w = Pr^{-(2/3)}\left(h_{aw} - h_w\right)\rho_e u_e\left(C_f/2\right) \tag{68}$$

Comparisons of this method of analysis with experimental wind-tunnel data has been made by Cresci, MacKenzie, and Libby (Ref. 57)

and by Nardo and Sadler (Ref. 59). These comparisons were based upon the Nusselt number evaluated from the expression

$$Nu = Pr^{1/3} \left(\frac{h_{aw} - h_w}{h_{s_e} - h_w} \right) \frac{\bar{\rho}\,\bar{u}}{\sigma_s \mu_e^{1/2}} \left(C_f/2 \right) Re_s \tag{69}$$

Figures 20(a) through 20(e) show the variation of Nusselt number along a meridian of the hemisphere used in the wind-tunnel tests. These results were obtained for $\tilde{R}e$ from 0.97×10^6 to 5.1×10^6 with the total temperature variation from approximately 1200—1900°R. The tunnel Mach numbers were not given. These results indicate a qualitative agreement of the computational procedure with the experimental results. It is readily apparent, however, that appropriate free-flight transition heat-transfer data will be required before the method can be used with confidence for design purposes. No such flight data are presently available in the open literature.

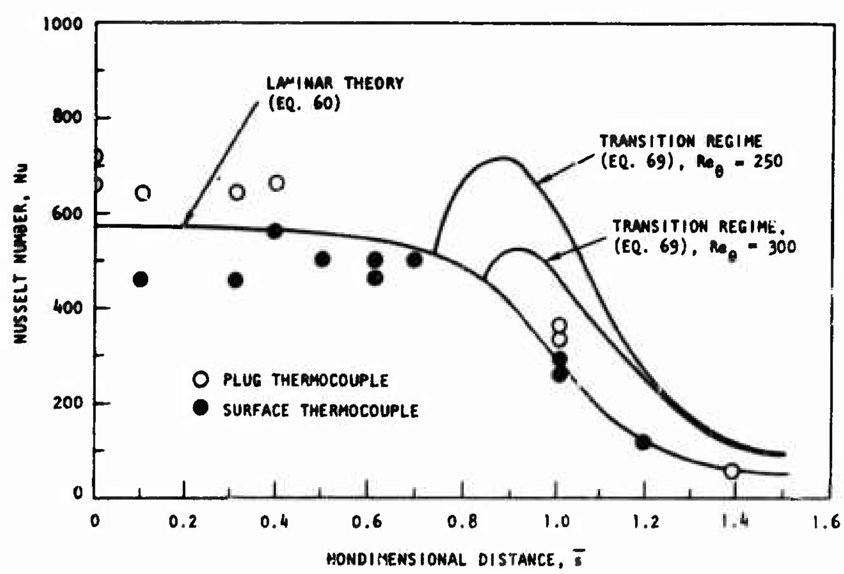
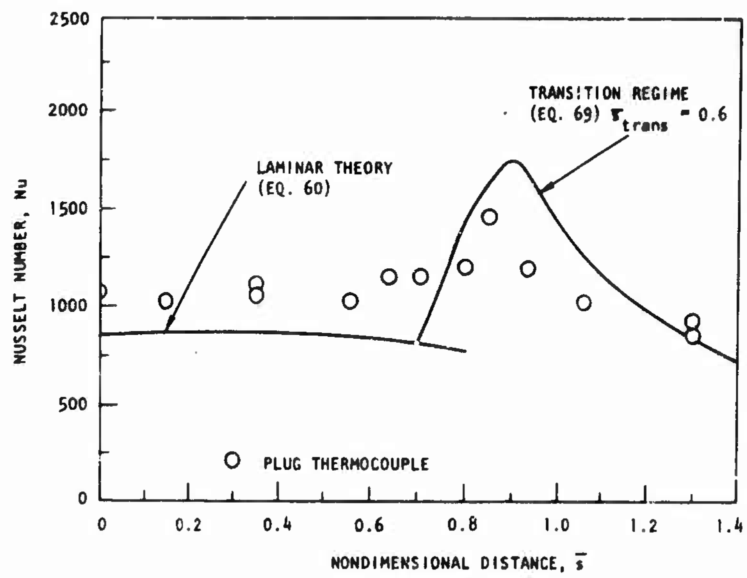
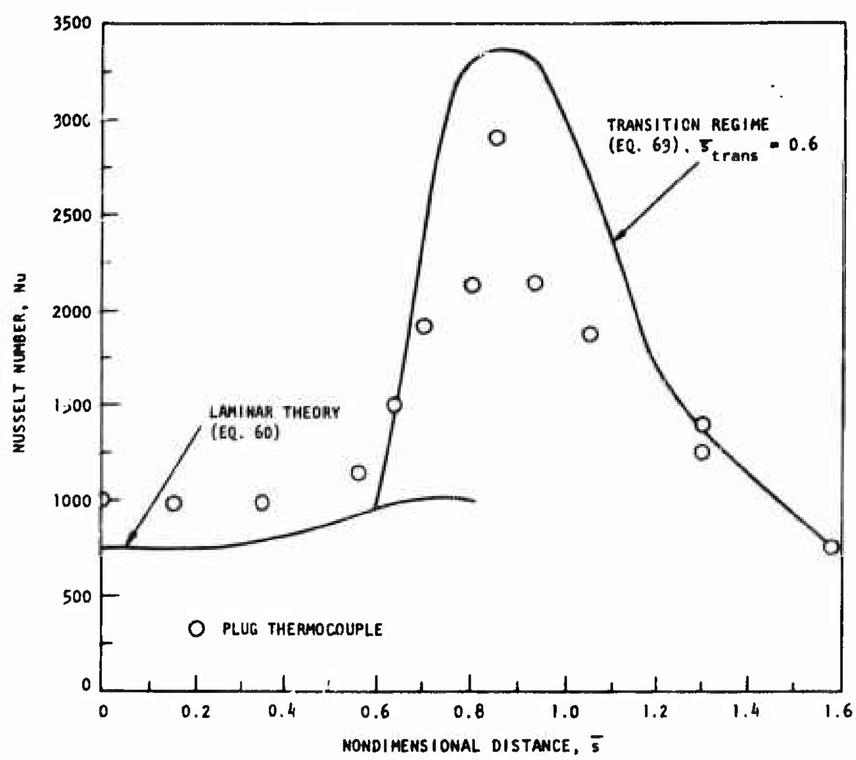


FIG. 20(a). Variation of Nusselt Number Along Meridian of Hemisphere, $\tilde{R}e = 0.97 \times 10^6$; Nardo and Sadler, Ref. 59.

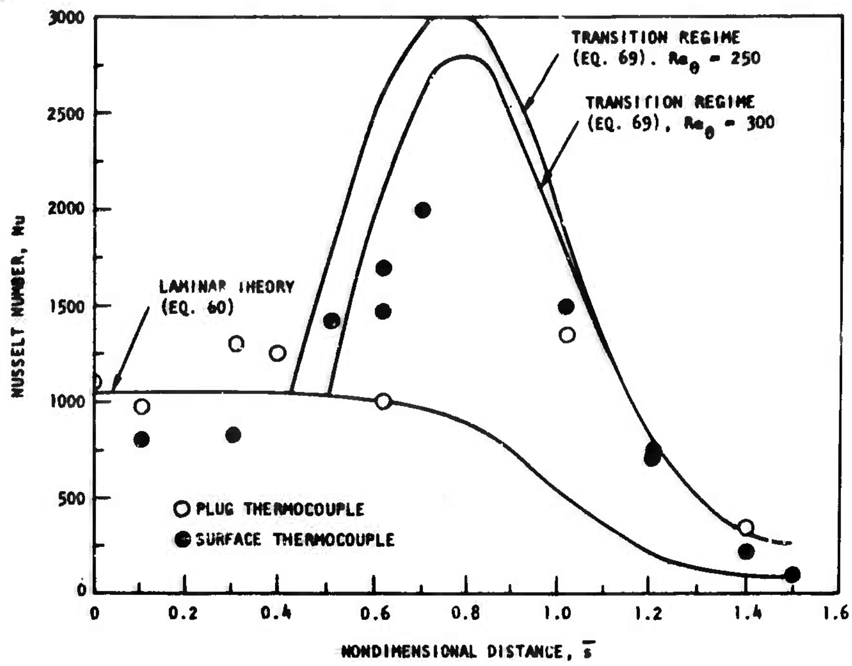


(b) $\tilde{Re} = 1.7 \times 10^6$

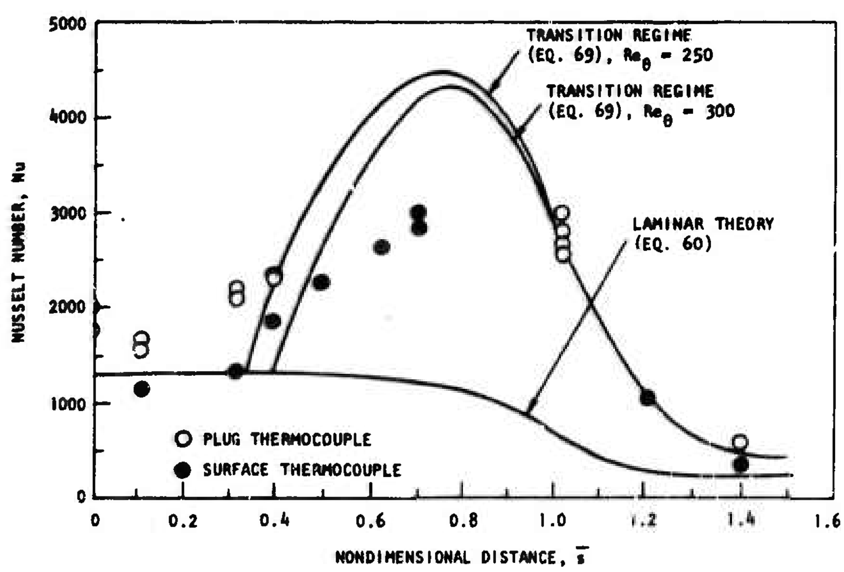


(c) $\tilde{Re} = 3.0 \times 10^6$

FIG. 20. Variation of Nusselt Number Along Meridian of Hemispherically Capped Cone; Cresci, MacKenzie, and Libby, Ref. 57.



(d) $\tilde{Re} = 3.3 \times 10^6$



(e) $\tilde{Re} = 5.1 \times 10^6$

FIG. 20. Variation of Nusselt Number Along Meridian of a Hemisphere; Nardo and Sadler, Ref. 59.

TURBULENT HEAT TRANSFER ON BLUNT BODIES OF REVOLUTION

Fully developed, turbulent heat transfer over blunt-body vehicles has been compared to the prediction of the flat-plate reference-enthalpy method by Cresci, MacKenzie, and Libby (Ref. 57) for wind-tunnel results, and by Zoby and Sullivan (Ref. 62), and by Eckert (Ref. 63) for the results of free-flight tests.

The correlations, presented by these authors, were made by relating the local skin-friction coefficient to the heating rate by a modified Reynolds analogy expressed as

$$C_H = \frac{\dot{q}_w}{\left[\rho_e u_e \left(h_{aw} - h_w\right)\right]} = \frac{C_f}{2} Pr^{-(2/3)} \tag{70}$$

To evaluate the Stanton number, C_H, for turbulent heat transfer, the local skin-friction coefficient, C_f, must be properly calculated. The Blasius relation for the incompressible skin-friction coefficient is

$$\left(\frac{C_f}{2}\right)_i = 0.0296 \left(Re_s\right)^{-0.2} \tag{71}$$

for the Reynolds number range $10^5 \leq Re_s \leq 10^9$, and the Schultz-Grunow incompressible skin-friction relation is

$$\left(\frac{C_f}{2}\right)_i = 0.185 \left(\log_{10} Re_s\right)^{-2.584} \tag{72}$$

for the Reynolds number range $10^5 \leq Re_s \leq 10^9$. Zoby and Sullivan (Ref 62) point out that both relationships correlate the data below a Reynolds number, Re_s, of 10^7 but that the Schultz-Grunow relation applies over the entire range of Reynolds numbers from 10^5 to 10^9.

Compressibility effects were accounted for by again evaluating the flow properties at reference conditions. The local skin-friction coefficient is then written

$$C_f/2 = 0.0296 \left(\rho^*/\rho_e\right)^{0.8} \left(\mu^*/\mu_e\right)^{0.2} \left(Re_s^*\right)^{-0.2} \tag{73}$$

and

$$C_f/2 = 0.185 \left(\rho^*/\rho_e\right) \left(\log_{10} Re_s^*\right)^{-2.584} \tag{74}$$

for the Blasius and Schultz-Grunow relations, respectively.

By substituting these expressions into Eq. 70, the following expressions for the heating rate are obtained:

$$\dot{q}_w = 0.0296\ \rho_e u_e \left(h_{aw} - h_w\right) \left(Pr^*\right)^{-2/3} \left(Re_s\right)^{-0.2} \left(\rho^*/\rho_e\right)^{0.8} \left(\mu^*/\mu_e\right)^{0.2} \tag{75}$$

and

$$\dot{q}_w = 0.185\ \mu_e u_e \left(h_{aw} - h_w\right) \left(Pr^*\right)^{-2/3} \left(\log_{10} Re_s^*\right)^{-2.584} \left(\rho^*/\rho_e\right) \tag{76}$$

It is easily shown that Eq. 70 may be written in terms of the Eckert flat-plate reference-enthalpy formula. If

$$Re_s^* = \frac{\rho^* u_e s}{\mu^*} \tag{77}$$

then

$$C_H^* = \frac{\dot{q}_w}{\rho^* u_e \left(h_{aw} - h_w\right)} = 0.0296 \left(Pr^*\right)^{-2/3} \left(Re_s^*\right)^{-0.2} \tag{78}$$

The Reynolds number is based upon properties evaluated at reference conditions, the velocity at the outer edge of the boundary layer, and the corresponding flat-plate wetted length.

Experimental results for turbulent heat transfer to blunt bodies of revolution obtained in wind tunnel tests, presented in Ref. 57, are shown in Fig. 21. These data were obtained from the shrouded-model wind-tunnel technique for tunnel total temperature of 1200—1900°R, hence, tunnel Mach numbers were not reported.

Shown also in Fig. 21 are results obtained by Buglia (Ref. 50) from free-flight tests of the smooth hemispherically capped cone over a Mach number range from 2.32 to 3.14.

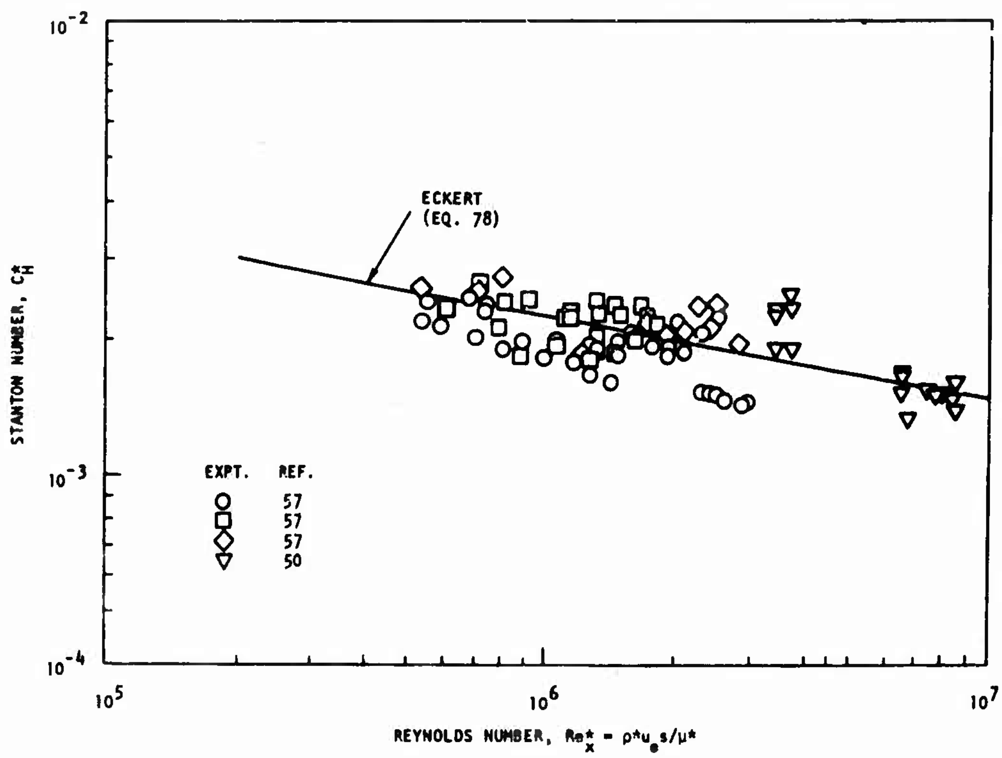


FIG. 21. Variation of Local Stanton Number Based on the Flat-Plate Reference-Enthalpy Method With the Local Reynolds Number; Cresci, MacKenzie, and Libby, Ref. 57; and Buglia, Ref. 50.

The correlation study of Zoby and Sullivan (Ref. 62) includes some turbulent heat-transfer data. These data cover a Mach number range from 2.98 to 5.3 and include some 13 data points. It should be pointed out that blunt-body, turbulent heat-transfer data are relatively scarce as compared to sharp cone results, and that additional turbulent heat-transfer results are needed below a Mach number of 3 to obtain more confidence in the engineering heat-transfer computational methods.

Eckert (Ref. 63) has presented a correlation of experimental results for turbulent flow over flat plates as a function of Mach numbers in the range from 0 to approximately 5. The experimental results are not pertinent to this report because they were obtained for a flat plate, but the trend of the limited data is important. Eckert found that the ratio of the actual to the constant-property turbulent-friction factor for a plate at the recovery temperature scattered rather widely from the theoretical values as the flow Mach number decreased from approximately 2 to approximately unity. If this trend is typical of heat-transfer results in this Mach number range, then the method of analysis for blunt-body heat transfer in the turbulent regime may be in question. Hence, even though satisfactory agreement for fully developed turbulent heat-transfer results may be obtained from the flat-plate reference-enthalpy method for Mach numbers greater than 1.6, one must be cautious in arbitrarily extrapolating the results to Mach numbers close to unity. Clearly, additional experimental results are required in this range.

For rapid calculation of turbulent skin-friction and heat-transfer results for high Mach number vehicles, Knuth (Ref. 64), and Neal and Bertram (Ref. 65) have presented turbulent skin-friction and heat-transfer charts adapted from the Spalding and Chi (Ref. 66) method for determining the drag of a compressible, turbulent boundary layer on a smooth flat plate. This method is presented for a Mach number range from 0 to 20, considering gross effects; hence, its usefulness in a limited range of Mach numbers from 0.7 to 5 where detail is important becomes questionable.

METHOD OF COMPUTATION OF THE HEAT-TRANSFER DISTRIBUTION

The calculation of the stagnation-point heat-transfer rate and the distribution of heat-transfer rate over the surface of a blunt-body vehicle are preferably performed by a digital computer. The slide rule can be used for initial estimates, but the computations can become tedious. The following discussion describes the method of calculation and the assumptions made.

The geometrical configuration is assumed to be a spherically capped frustrum of a cone.

Effects of wall temperature, Mach numbers, and variable fluid properties are accounted for by evaluating the fluid properties at the appropriate reference conditions. Correspondingly, the theoretical variation of Stanton number versus Reynolds number, based on the reference-enthalpy method, is a straight line on a log-log graph, either for laminar flow, Eq. 49, or turbulent flow, Eq. 78. The state of the boundary layer may be readily ascertained from such a plot, as indicated in Fig. 18 (a—f) through 20(a—e).

To summarize the method, the flow at the outer edge of the boundary layer is assumed to have passed through a normal shock wave, followed by isentropic expansion to the theoretical value of the surface pressure found from Eq. 12. Fluid properties are evaluated at the reference condition, thereby eliminating the dependence of Stanton number and Reynolds number on Mach number, wall temperature, and variable fluid properties. Air properties are taken from Keenan and Kayes "Gas Tables," whereas the viscosity is evaluated by using Sutherland's equation.

The input information required for the computation of heat-transfer distributions is assumed to be (1) vehicle nose radius, (2) flight velocity, (3) ambient temperature, (4) ambient density, and (5) wall temperature.

Stagnation-Point Heat-Transfer Rate

The calculation of the stagnation-point heat-transfer rate proceeds as follows:

1. Compute the flight Mach number, M_∞, from the flight velocity, V_∞(fps), and the ambient air temperature, T_∞(°R):

$$M_\infty = \frac{V_\infty}{49.1\sqrt{T_\infty}}$$

2. Compute the pressure ratio, P_{s_e}/P_∞, across the normal shock wave from the flight Mach number M_∞:

$$\frac{P_{s_e}}{P_\infty} = \frac{\left\{[(\gamma+1)/2]\ M_\infty^2\right\}^{\gamma/\gamma-1}}{\left[2\gamma/(\gamma+1)\ M_\infty^2 - (\gamma-1)/(\gamma+1)\right]^{1/\gamma-1}}$$

For $\gamma = 1.40$, this equation reduces to

$$\frac{P_{s_e}}{P_\infty} = 1.2\ M_\infty^2 \left(\frac{7.2\ M_\infty^2}{7\ M_\infty^2 - 1}\right)^{2.5}$$

3. Compute the ratio of the total temperature at the outer edge of the boundary layer at the stagnation point to the free-stream static temperature:

$$\frac{T_{s_e}}{T_\infty} = 1 + \frac{\gamma - 1}{2} M_\infty^2$$

4. From the gas tables, extract the enthalpy, h_{aw}, corresponding to the temperature, T_{s_e} (°R), and the enthalpy, h_w, corresponding to the wall temperature, T_w (°R). For low temperatures, corresponding to $M_\infty < 5$, the specific heat at constant pressure may be assumed constant at a value of $c_p = 0.246$ Btu/lbm°R.

5. Compute the density at the edge of the boundary layer at the stagnation point, $\rho_{e_s} = P_{s_e}/\bar{R}\,T_{s_e}$ lbm/ft^3, and the density of air at the wall temperature at the nose of the vehicle

$$\rho_{w_s} = \frac{P_{s_e}}{\bar{R}T_{ws}} \frac{\text{lbm}}{\text{ft}^3}$$

where $\bar{R}$ is the gas constant for air.

6. Compute the viscosity of the air at the stagnation-point temperature, T_{s_e} (°R), and at the wall temperature at the nose of the vehicle, T_{w_s} (°R), from Sutherland's formula

$$\mu = \frac{2.270 \times 10^{-8}\,(T)^{3/2}}{T + 198.6} \quad \frac{\text{lb}_f\text{-sec}}{\text{ft}^2}$$

where T is the temperature in degrees Rankine.

7. Compute the stagnation-point heat-transfer parameter, $\dot{q}_{w_s} \sqrt{R_o/P_{s_e}}$ from the results of Fay and Riddell, Eq. 40:

$$\dot{q}_{w_s} \sqrt{R_o/P_{s_e}} = 0.763 \left(Pr^{-0.6}\right)\left(\rho_{w_s} \mu_{w_s}\right)^{0.1}\left(\rho_{e_s} \mu_{e_s}\right)^{0.4}\left(h_{aw} - h_w\right)$$

$$\left[\left(2\bar{R}T_{s_e}\right)\left(1 - \frac{P_\infty}{P_{s_e}}\right)\right]^{\frac{1}{4}} \frac{1}{\sqrt{P_{s_e}}}$$

For a given nose radius, R_o, this result then yields the stagnation-point heat-transfer rate.

Heat-Transfer Distribution Over Hemisphere Surface

To compute the heat-transfer distribution over the hemisphere surface, the expression of Lees, Eq. 50 and 51 may be used as follows:

1. Compute the heat transfer on the hemispherical cap at the angle θ from Lees relationship, Eq. 50 and 51. The angle θ is the angle between the flight direction and the radius vector to the surface station of interest. The stagnation-point heat-transfer rate is obtained from the previous calculation.

An alternate method for computing the local value of the heat transfer employs the Korobkin expression for the Stanton number, Eq. 49. The procedure for obtaining the heat flux from this expression is as follows:

2. Compute the pressure, P_e, at the edge of the boundary layer at the point θ from Eq. 12 for the pressure coefficient. The Newtonian pressure distribution may be used to advantage over the surface of the hemisphere.

3. Compute the static temperature, T_e, at the outer edge of the boundary layer, assuming isentropic flow from the stagnation point, from

$$\frac{T_e}{T_{s_e}} = \left(\frac{P_e}{P_{s_e}}\right)^{0.286}$$

4. Compute the velocity, u_e(fps), of the flow at the outer edge of the boundary layer from the flight velocity, V_∞(fps), and the enthalpies, h_e(Btu/lbm) and h_∞(Btu/lbm), corresponding to the temperatures T_e and T_∞.

$$u_e = \sqrt{V_\infty^2 + 50{,}100\left(h_\infty - h_e\right)}$$

5. Compute the Reynolds number corresponding to the local flow conditions at the outer edge of the boundary layer at the distance, s(ft),

$$Re_s = \frac{\rho_e u_e s}{\mu_e}$$

6. If $Re_s < 10^6$, assume laminar boundary layer. If $Re_s > 10^6$, assume turbulent boundary layer.

7. Compute a, the recovery factor, r^*; b, the adiabatic wall enthalpy; and c, the reference temperature, T^*(°R):

a. Recovery factor

$$r^* = \begin{cases} Pr^{*1/2} & \text{for laminar flow} \\ Pr^{*1/3} & \text{for turbulent flow} \end{cases}$$

b. Adiabatic wall enthalpy

$$h_{a_w} = h_e + r^* \frac{u_e^2}{50{,}100} \quad \frac{\text{Btu}}{\text{lbm}}$$

c. Reference temperature

$$T^* = T_e + 0.50\left(T_w - T_e\right) + 0.22\left(T_{aw} - T_e\right)$$

where the adiabatic recovery temperature, T_{aw}, corresponds to the adiabatic wall enthalpy.

8. Compute the density, $\rho^*(\mathrm{lbm/ft^3})$, and viscosity, $\mu^*(\mathrm{lb_f\text{-}sec/ft^2})$, corresponding to the local pressure, $P_e(\mathrm{lb_f/ft^2})$ and reference temperature, $T^*(°\mathrm{R})$.

9. Compute the Reynolds number based on reference conditions and the running length along the surface from the stagnation point to the given station:

$$Re_s^* = \frac{\rho^* u_e s}{\mu^*}$$

10. For laminar flow on the hemisphere surface, use Korobkin's equation, Eq. 49, for the Stanton number on a sphere:

$$C_H^* = 0.763 \left(Pr^*\right)^{-0.6} \left(Re_s^*\right)^{-1/2}$$

11. The heat flux at the station of interest is then computed from

$$\dot{q}_w = C_H^* \rho^* u_e \left(h_{aw} - h_w\right)$$

12. For turbulent flow, the Eckert flat-plate equation may be used to compute the Stanton number:

$$C_H^* = 0.0296 \left(Pr^*\right)^{-2/3} \left(Re_s^*\right)^{-0.2} \quad \text{for } 10^5 \leq Re_s \leq 10^7 \; .$$

13. When a wider range of Reynolds number, Re_s, is encountered for turbulent flow, the Schultz-Grunow relation may be used:

$$C_H^* = 0.185 \left(Pr^*\right)^{-2/3} \left(\log_{10} Re_s^*\right)^{-2.584}$$

Heat Transfer in the Transitional Region

The procedure for the determination of the heat transfer in the transitional region between the laminar and turbulent heat-transfer regions is outlined as follows:

1. The Reynolds number, Re_θ, based upon the momentum thickness may be chosen as the parameter to represent transition. This momentum Reynolds number, Re_θ, based on local fluid properties at the outer edge of the boundary layer and the laminar boundary layer momentum thickness, may be computed from Ref. 57:

$$Re_\theta = \frac{0.22}{C_f/2} = 0.22 \left(Pr_e\right)^{-1} \left(C_{H_e}\right)^{-1}$$

Transition is assumed to occur at $Re_\theta = 250$. Highly polished surfaces will delay transition; hence, higher values of Re_θ for transition will occur. These values will have to be determined experimentally.

2. The constant B is evaluated from the laminar flow such that the local friction coefficient, $C_f/2$, remains continuous from the laminar to turbulent transition region:

$$\Pi = \left[Re_\theta^{\,2} \left(\frac{0.013}{Re_\theta^{1/4}} - \frac{C_f}{2} \frac{\mu_{s_e}}{\mu_e} \right) \right] \text{ at transition}$$

3. Calculation of the skin-friction and the momentum Reynolds number through the transition region proceeds through the numerical solution of the following equations:

$$\frac{C_f}{2} = \frac{\mu_e}{\mu_{s_e}} \left(\frac{0.013}{Re_\theta^{1/4}} - \frac{\Pi}{Re_\theta^{\,2}} \right)$$

$$\frac{dRe_\theta}{d\bar{s}} = Re \left(\frac{\rho_e u_e}{\mu_e} \right) \left(\frac{\mu_{s_e}}{\rho_{s_e} \sqrt{h_{s_e}}} \right) \left(\frac{C_f}{2} \right)$$

$$- Re_\theta \, d\left\{ \ln \left[\left(r_o/R_o \right) \mu_e/\mu_{s_e} \right] \right\} / d\bar{s}$$

4. The heat-transfer distribution is then calculated from

$$\dot{q}_w = \left(C_f/2\right) \left(Pr_e\right)^{-2/3} \rho_e u_e \left(h_{aw} - h_w\right)$$

5. Finally, the local Nusselt number is calculated from

$$Nu = Pr^{1/3} \left(\frac{h_{aw} - h_w}{h_{s_e} - h_w}\right) \frac{\left(\rho_e/\rho_{s_e}\right)\left(u_e/u_{s_e}\right)}{\left(\sigma_{s_e}\right)}$$

where

$$Re = \frac{\rho_{s_e} u_\infty R_o}{\mu_{s_e}}$$

is the Reynolds number based on flight velocity, stagnation properties, and the radius of the hemisphere.

CONCLUSIONS

PRESSURE DISTRIBUTIONS

In addition to a literature survey, methods for determining the pressure distribution on bodies of revolution in flight regimes from high subsonic through supersonic are discussed in this report as stated in the Introduction. From this investigation, the modified Newtonian and the modified Newtonian/shock-expansion methods give good accuracy with a minimum of effort for the following cases:

1. For a hemisphere-cylinder, the pressure distribution may be determined from either Eq. 3 or 13 in combination with Eq. 12 for the Mach number range from 0.75 to 2.0, and from Eq. 2 and 3 in combination with the shock-expansion method for Mach numbers above 2.

2. For other blunt bodies of revolution that have continuous curvature in the nose region, Eq. 15 is used for Mach numbers from 0.7 to 2.0, and Eq. 2 is used for Mach numbers above 2.

3. For pointed bodies, Eq. 6 and the modified Newtonian/shock-expansion method are used for Mach numbers above 1.0.

4. Equation 11 may be used for cylindrical afterbodies.

For pointed bodies at Mach numbers other than those specified as being ammenable to the modified Newtonian/shock-expansion method, the most useful techniques are those of the small-perturbation theory as developed by Van Dyke and Spreiter.

HEAT-TRANSFER DISTRIBUTIONS

This report gives an evaluation of applicable blunt-body heat-transfer rates for high-subsonic to low-supersonic flight. An outline of an engineering method of analysis for the prediction of blunt-body heat transfer within this flight regime is also presented. From this study, the following conclusions can be made.

1. Stagnation-point heat transfer to blunt-body vehicles for a range of flight Mach numbers from 2.3 to 3.1 has been predicted by the formula of Fay and Riddell with an accuracy from 8 to 20%, inclusively. However, the accuracy of prediction of wind tunnel data by the formula of Fay and Riddell is dependent upon the enthalpy difference across the boundary layer and not upon the tunnel Mach number. Thus, experimental data are required to determine the applicability of the formula of Fay and Riddell for Mach number regions of less than 2.

2. The laminar heat-transfer distributions over blunt bodies of revolution at Mach numbers between approximately 2 and 5 are best represented by Lees' method. If laminar flow persists to the juncture of hemisphere and cone on a hemispherically capped cone, Lees' method will predict the effect of pressure gradient on the reduction of the heat transfer. However, for the laminar region between the stagnation point and the body station at 45 degrees, little difference is noted between Korobkin's method and Lees' method.

3. The method of predicting the heat-transfer rate in the transition region on blunt-body vehicles developed by Cresci, MacKenzie, and Libby; Nardo and Sadler; and Stetson is apparently a rational and fairly accurate method for including transitional behavior in heat-transfer calculations. Nevertheless, methods for predicting transition still remain to be developed.

4. Fully developed, turbulent heat-transfer rates over blunt-body vehicles for a Mach number range from 1.6 to above 5 are adequately represented by Eckert's flat-plate reference enthalpy technique. No

turbulent heat-transfer data are available in the open literature for either flat plates or for blunt-body systems in the Mach number range from 0.7 to 1.6. Such data will be required to provide confidence in the application of the engineering methods discussed for this range of Mach numbers. Extrapolation of the flat-plate formulas into this range of Mach numbers may be of questionable accuracy.

Appendix

BIBLIOGRAPHY

This bibliography consists of three sections. The first section is an annotated bibliography that cites and critically reviews 51 entries of particular interest in the determination of pressure and heat-transfer distributions on bodies of revolution for subsonic, transonic, and supersonic flight regimes. The second section, a general bibliography, cites 516 entries of interest in this area. Some additional references covering hypersonic flow, boundary-layer theory, real gas effects, and corrections for angle of attack or yaw, which contain pertinent information, are also included. This overall bibliography is not exhaustive, but it is extensive and does present the major portion of those articles having a direct bearing in the development of engineering methods of computing pressure and heat-transfer distribution.

The annotated bibliography contains entries arranged alphabetically by author. Those entries covering pressure distribution are presented first (Ref. 1– 23), followed by those covering heat-transfer distribution (Ref. 24—51). The general bibliography contains entries, again, arranged alphabetically by author; however, these are separated into six main groups by subject. The largest groups, those covering pressure distribution and heat-transfer distribution, are further subdivided for more convenient use.

ANNOTATED BIBLIOGRAPHY

PRESSURE DISTRIBUTION

1. Adams, M. C. and Sears, W. R., "Slender-Body Theory--Extensions and Review," Journal of Aeronautical Sciences, Vol. 20, No. 2, February 1953, pp. 85-98. (see also, Sears, W. R., "Small Perturbation Theory," High Speed Aerodynamic and Jet Propulsion, Vol. 6, Part C., Princeton University Press, Princeton, New Jersey, 1954.

   A review is presented of small perturbation theory as applied to wings and slender bodies of revolution. Consideration is given to determination of pressure distribution from theoretical developments which are obtained by linearized methods, similarity, and simple corrections to incompressible distributions. This report provides a good introduction to the subject and suggests several useful modifications of earlier methods which give improved accuracy. A number of comparisons of theoretical and test results are given. The test data, however, has been drawn from other sources.

2. Andrews, J. S., "Steady State Airload Distribution on a Hammerhead-Shaped Payload of a Multistaged Vehicle at Transonic Speeds," D2-22947-1, The Boeing Company, Seattle, Washington, February 1964.

   Data is presented for pressure distribution on a hemisphere cylinder for a Mach number range from 0.8 to 2.0 (see Table 1A for data summary). Additional and empirical modifications of the modified Newtonian theory is developed which correlates the data for $0.75 \leq M \leq 2.0$.

   $$C_p = C_{p_{max}} \sin^2\theta_b + F(\theta_b, M_\infty)$$

   $$F(\theta_b, M_\infty) = [0.78(\frac{1}{M_\infty})^{2.27} \sin^2\theta_b - 0.95e^{-2.235(M_\infty - 1)}]\cos\theta_b$$

3. Baer, A. L., "Pressure Distribution on a Hemisphere Cylinder at Supersonic Mach Numbers," AEDC-TN-61-96, August 1961 (AD 261 501).

   Experimental pressure distributions were obtained over the hemispherical nose of AGARD Model E at Mach numbers ranging from 2 to 8 (see Table 1A for summary of data). Tests results are compared with modified Newtonian theory and show good agreement. Comparison with the numerical methods of Van Dyke and Gordon (NASA TR R-1, 1959) are also given.

4. Blick, E. F. and Francis, J. E., "Spherically Blunted Cone Pressure Distribution," AIAA Journal, Vol. 4, No. 3, March 1966, pp. 547-549.

   The modified Newtonian-shock expansion method is applied to the determination of the pressure distribution on spherically blunted cones. The angle at which the modified Newtonian and the shock-expansion methods match is given by the empirical equation

$$\delta_m = 0.349 + [0.198 + 0.80(M_\infty - 2.8)^{-0.455}]$$

$$\text{arc cos } [1.05/\ln M_\infty)]\text{rad}$$

for the range $3 \leq M_\infty \leq 15$.

5. Cole, J. D., "Newtonian Flow Theory for Slender Bodies," Journal of Aeronautical Sciences, Vol. 24, No. 6, June 1957, pp. 448-455.

   Newtonian flow theory is examined from the point of view of gas dynamics and hypersonic small-disturbance theory. A general solution of the first approximation for the flow past slender bodies at zero angle of attack is given.

   This paper is one of the earlier efforts in application of Newtonian theory and its major concern is the hypersonic flow regime.

6. Guderley, G. and Yoshihara, H., "Axial-Symmetric Transonic Flow Patterns." USAF Technical Report 5797. USAFAMC, Dayton, Ohio, 1949.

   A solution for the flow field about an axisymmetric body at free stream Mach number one is presented. The method is of the indirect type, working from an assumed potential flow field to the resulting description of a nose shape. This approach appears to be too complicated to be useful in terms of an "engineering" solution for pressure distribution on a specific body.

7. Katz, J. R., "Pressure and Wave Drag Coefficients for Hemisphere, Hemisphere Cones, and Hemisphere Ogives," NAVORD Report 5849, March 1958.

   Experimental pressure coefficients for hemisphere cylinders and hemisphere cones are presented and compared with modified Newtonian theory (a summary of the data is presented in Table 1A). No data is presented for pressure distribution on hemisphere ogives. The results show good correlation of experimental data for a Mach number range 2.5 to 6.5 with the modified Newtonian form

$$C_p = C_{p_{max}} \sin^2\theta_b$$

8. Laitone, E. V., "Linearized Subsonic and Supersonic Flow about Inclined Slender Bodies of Revolution," Journal of Aeronautical Sciences, Vol. 14, No. 11, November 1947, pp. 631-642.

   Rigorous first-order linearized solutions are obtained for subsonic and supersonic flow about bodies of revolution. The surface pressure is found by means of a Taylor's Series Expansion in terms of the cross-sectional area. The resulting expressions are in satisfactory agreement with existing experimental data. The first order solution for the subsonic regime is strictly valid only for incompressible flow or for a slender pointed body in the compressible subsonic regime. However, except in the region of a stagnation point, the solution is also a valid approximation for

slender bodies with blunt ends. The solution for the supersonic regime is only valid where the entire flow field is supersonic. This imposes the requirement that the nose be pointed and the shock wave attached.

Some comment on the pressure distribution at small angles of attack is also made.

9. Laitone, E. V., "The Subsonic Flow about a Body of Revolution," Quarterly Applied Mathematics, Vol. 5, Part 2, July 1947.

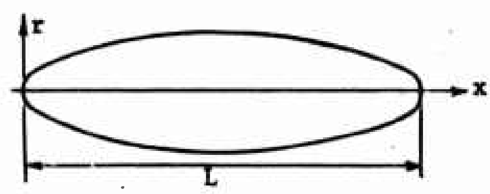
A solution is presented for the linearized flow potential around axisymmetric bodies of revolution in the subsonic regime:

$$\phi_{xx} + \phi_{rr} + 1/r\ \phi_r = 0$$

This is accomplished by approximating the source-sink distribution integral by a Taylor Series Expansion. This process yields the following first-order approximation for the pressure distribution

$$C_p = + \frac{1}{\pi}\frac{s'}{2x}\left(\frac{1-2x/d}{1-x/L}\right) - s''\left(1 + \ln\frac{\beta r}{L} - \ln 2\sqrt{\frac{x}{L}\left(1 - \frac{x}{L}\right)}\ \right)$$
$$+ \frac{s^{(3)}}{4} L\left(1 - \frac{2x}{L}0 + \frac{s^{(4)}}{24} L^2\left(1 - \frac{2x}{L} + \frac{2x^2}{L^2}\right) + \dots - (r')^2$$

where the coordinates are as follows



The values of C_p obtained are strictly valid only for incompressible flow or for pointed slender bodies in subsonic compressible flow. However, except in the region of a stagnation point, the expression offers a valid approximation for the pressure distribution on blunt slender bodies of revolution.

10. Lees, L., "Hypersonic Flow," Proceedings of the Fifth International Aeronautical Conference, IAS-RAES, June 20-23, 1955.

A considerable contribution to the usefulness of Newtonian theory is made by suggesting a modified form for the pressure coefficient on blunt bodies

$$C_p = C_{p_{max}} \sin^2\theta_b$$

where $C_{p_{max}}$ = the value at the stagnation point.

11. Love, E. S., "Generalized Newtonian Theory," Journal of Aerospace Sciences, Vol. 26, No. 5, May 1959, p. 314.

A means of applying modified Newtonian theory to pointed bodies of revolution is given. Using Lees' form

$$C_p = C_{p_{max}} \sin^2\theta_b$$

$C_{p_{max}}$ is defined as

$$C_{p_{max}} = C_{p_N}/\sin^2\theta_N$$

where

$$C_{p_N} \equiv C_p \text{ at nose of cone with semi-vertex angle } \theta_N$$

$$\theta_N \equiv \text{local slope at pointed nose of the body}$$

This application is, however, limited to the portion of the supersonic flight regime where the body will have an attached shock at the nose.

12. Matthews, C. W., "Comparison of the Experimental Subsonic Pressure Distribution about Several Bodies of Revolution with Pressure Distribution Computed by Means of the Linearized Theory," NACA TN 2519, February 1952.

Experimental pressure distributions on two prolate spheroids of fineness ratios 6 and 10, an ogive body and a prolate spheroid with an angular bump are compared with theoretical results obtained from applying the Prandtl-Glauert correction to the incompressible potential flow equations and from linearized theory. It is concluded that the rather more complicated Prandtl-Glauert method gives satisfactory results except in the region immediately adjacent to the ends. It is also concluded that the linearized theory does not yield satisfactory results. However, it should be noted that in arriving at this conclusion, Matthews used an incomplete expression for C_p. Later in reviewing this work Adams and Sears point out that using the complete expression for C_p

$$C_p = -2\phi_x - \phi_r^2$$

linearized theory predicts the experimental results of this report equally as well as the Prandtl-Glauert convection method and with less effort.

13. Okauchi, K., "Summary of Results of Pressure Coefficient Calculations for a Spherical-Tipped Tangent-Ogive Body," NAVWEPS Report 8048, October 1962.

A review of results obtained in work conducted by the Naval Ordnance Test Station to develop procedures for calculating pressure on practical body shapes is presented.

The body used for the test had a spherically-tipped (r_c = 0.094 in.) tangent ogive nose (r_b = 0.592 in.). Comparison of test results and theoretical results obtained from Van Dyke's second-order method is made at M_∞ =

1.53 and 2.17. In the theoretical analysis it is assumed that the body is pointed. The agreement is quite good (away from the nose tip). For M_∞ = 3.24 and 4.84 a method of characteristic solution is compared with test results. Again the body is assumed to be pointed in the theoretical calculations and the results showed good agreement.

This report also contained data for M_∞ = 0.75, 0.85, 0.95, 1.05, and 1.15. There is no attempt to present theoretical comparisons for these Mach numbers.

14. Perry, J. C. and Pasiuk, C., "A Comparison of Solutions to a Blunt Body Problem," AIAA Journal, Vol. 4, No. 8, August 1966, p. 1438.

Consideration is given to the various numerical methods of predicting the flow in the nose region of a blunt body moving at supersonic speeds. Fourteen different solutions were obtained and compared with experimental results for an elipsoide of revolution ($r = \sqrt{2x - 4x^2}$) at M_∞ = 3. This comparison showed several numerical methods closely predicted the measured test results. There was scatter between the pressures predicted by various numerical solutions, however. This scatter was considerable in the nose region.

The numerical methods used are based on one or the other of three different computational approaches developed by Belotserkorski, Van Dyke, and Swigart. A method developed at NASA by Lomax and Inouge using Van Dyke's approach and a method using Belotserkorski's approach developed at NWL by DiDonato provided quite good results.

15. Spreiter, J. R., "Aerodynamics of Wings and Bodies at Transonic Speeds," Journal of the Aerospace Sciences, Vol. 26, No. 8, August 1959, pp. 465-486.

A brief summary is presented on basic concepts and the principle results that have emerged from numerous experimental and theoretical investigations of transonic flow past thin wings and slender bodies. Emphasis is placed throughout on the correlation and evaluation of resulted provided by diverse methods for the approximate solutions of the nonlinear equations of the small perturbation theory in transonic flow. Also a critical examination of experimental results is given.

This paper is particularly valuable as a review of material pertinent to transonic flight. However, the bulk of the material presented is for two-dimensional bodies or pointed bodies of revolution and also appears in Spreiter and Alksne (NASA TR R-2) (see Table 1a for data summary).

16. Spreiter, J. R. and Alksne, A. Y., "Slender Body Theory Based on Approximate Solution of the Transonic Flow Equation," NASA TR R-2, 1959.

Approximate solutions of the nonlinear equations of the small perturbation theory for transonic flow are found for pointed slender bodies of revolution for flows with free-stream Mach number 1 and for flows that are either purely subsonic or supersonic. These results are obtained by application of a method based on local linearization. The theory is developed for pointed bodies of arbitrary shape and specific results are given for

cone cylinders and parabolic arc bodies at zero angle of attack. Results are compared with data (see Table 1A for data summary).

The method of this report is rather complicated but does provide solutions for the pressure distribution on pointed bodies in transonic flight not found elsewhere. Additionally, the report is of considerable value as general background material.

17. Van Dyke, M. D., "The Paraboloid of Revolution Subsonic Flow," Journal of Mathematics and Physics, Vol. 37, No. 1, January 1958, pp. 38-51.

The solution for the flow about a paraboloid of revolution is developed and it is demonstrated how this solution may be applied to remedy the defects which arise when linearized small perturbation theory and its higher approximations are applied to bodies having round noses. This report provides useful background when applying the methods developed by Van Dyke in NASA TR R-47.

18. Van Dyke, M. D., "Practical Calculation of Second-Order Supersonic Flow Past Nonlifting Bodies of Revolution," NACA TN 2744, July 1952.

Calculation of second-order supersonic flow past pointed bodies of revolution at zero angle of attack is described in detail and reduced to routine computation. Tables of basic functions and standard computation forms are presented. The procedure is summarized so that one can apply it without necessarily understanding the details of the theory. A sample calculation is also given.

19. Van Dyke, M. D., "Second-Order Slender Body Theory--Axisymmetric Flow," NASA TR R-47. 1959.

Small perturbation theory for subsonic and supersonic flow past bodies of revolution is developed. Methods are developed for handling the difficulties which arise at round ends in subsonic flow. In addition to second-order theory, first order linearized theory and the transonic approximation to the non-linear equations are developed and discussed in some detail.

The methods presented are somewhat difficult in application but the report presents a very good comprehensive review of perturbation theory. All results obtained are compared with other existing theories or with experimental data (see Table 1A for data summary).

20. Van Dyke, M. D., "The Similarity Rules for Second-Order Subsonic and Supersonic Flow," NACA Report 1374, 1958.

The similarity rules for linearized compressible flow theory (Göthert's rule and its supersonic counterpart) are extended to second order. It is shown that any second-order subsonic flow can be related to "nearly incompressible" flow past the same body, which can be calculated by the Janzen-Rayleigh method. It is also pointed out that these methods are chiefly useful as a theoretical aid rather than in correlation of experimental data. This arises as tests on four related bodies would be needed to isolate the four functions involved in the similarity expression.

21. Van Dyke, M. D., "A Study of Hypersonic Small Disturbance Theory," NACA TN 3173, May 1954.

Newtonian theory is suggested as a practical means of determining the pressure distribution over bodies of revolution in hypersonic flight. For the present purpose the value of this report is chiefly historical in that it records the first application of Newtonian theory to problems of current interest. In its original form Newtonian theory yields a pressure coefficient of the form

$$C_p = 2 \sin^2\theta_b$$

22. Vendemia, R. J., "An Engineering Method for Rapid Calculation of Supersonic-Hypersonic Pressure Distributions on Lifting and Non-Lifting Pointed Bodies of Revolution and Several Special Cases of Blunt-Nosed Bodies of Revolution," TG 752, The Johns Hopkins University Applied Physics Laboratory, Silver Spring, Maryland, November 1965.

A method is presented for calculating pressure distribution and local Mach numbers for pointed bodies of revolution and for several cases of blunt-nosed bodies in the supersonic and hypersonic flight regimes. The method used is a hybrid solution involving modified Newtonian and shock-expansion theory. This approach is very useful and provides accurate results for a variety of nose shapes and fineness ratios in the supersonic to hypersonic regime. A considerable amount of test data is also presented and compared with theory (see Table 1A for a summary of Mach numbers and nose shapes). In the context of the present problem this report is of considerable value.

23. Wagner, R. D., Jr., "Some Aspects of Modified Newtonian and Prandtl-Meyer Expansion Method for Axisymmetric Blunt Bodies at Zero Angle of Attack," Journal of Aerospace Sciences, Vol. 26, No. 12, December 1959, p. 851.

A method is presented for determining the point at which the modified Newtonian and shock-expansion solutions should be matched. θ°_{MP} and M_{MP} are given in graphical form as a function of Mach number ($3 \leq M \leq 40$) for a hemisphere cylinder in air and helium.

HEAT-TRANSFER DISTRIBUTION

24. Beckwith, I. V. and Gallagher, J. J., "Heat Transfer and Recovery Temperatures on a Sphere with Laminar, Transitional, and Turbulent Boundary Layers at Mach Numbers of 2.00 and 4.15," NACA TN 4125, December 1957.

An investigation was made of the pressure, equilibrium, temperatures, and aerodynamic heat transfer on a sphere and on the nose of a hemisphere cylinder at Mach numbers of 2.00 and 4.15. The heat-transfer coefficients at the stagnation point from the tests at both Mach numbers were compared with the theory of Reshotko and Cohen (NACA TN 3513) and were in good agreement. The velocity gradients used in the theory were evaluated from the experimental pressure data.

The local heat-transfer coefficients from the tests at Mach number 2.00 were in agreement with the theory for laminar flow in the region up to approximately 20° from the stagnation point. Large increases in heat-transfer coefficient caused by transition to turbulent flow occurred beyond this region with the maximum values observed at about 40° from the stagnation point. From this location back to the separation point the approximate level in the heat-transfer coefficient as well as the variation with distance was fairly well predicted by a simple theory based on turbulent heat-transfer formulas for flat plates.

The heat transfer coefficients from the tests at Mach number 4.15 at the highest Reynolds numbers were generally similar to the results at Mach number 2.00 except that transition usually occurred farther back on the nose. The maximum values of the heat transfer coefficient downstream of transition were again in reasonably good agreement with the simple turbulent-flow theory. For Reynolds number of about 3.00 x 10^6 or less, the data over the entire hemispherical nose were in agreement with the theory for laminar flow.

25. Brindle, C. C. and Malia, M. J., Jr., "Longitudinal Aerodynamic and Heat Transfer Characteristics of a Hemisphere-Cylinder Missile Configuration with an Aerodynamic Spike," Aero Report No. 1061, Washington, D.C., The David Taylor Model Basin Aerodynamics Laboratory, AD No. 417 466, July 1963.

A hemisphere cylinder with an aerodynamic spike (produced by a small nozzle exhausting air from the hemispherical nose) was tested in the 18-inch wind tunnel at Mach numbers of 1.87 and 2.16. Heat-transfer coefficients were measured for both hot and cold bodies. No comparison or correlations with theory were obtained

26. Brunk, J. E. and White, W. N., "Free-Flight Hypersonic Heat Transfer and Boundary Layer Transition Studies: HTV Flights A-40 and A-41," ARL Technical Report 60-311, Wright-Patterson Air Force Base, Ohio, Aeronautical Research Laboratory, December 1960.

Two HTV-1 hypersonic test vehicles, Rounds A-40 and A-41, were flown at Holloman AFB in October 1959 with blunted and sharp 20° half angle nose cones, respectively. A maximum flight velocity of 5800 feet per second was attained, corresponding to a local sharp core Mach number and unit Reynolds number of 3.4 and 50 x 10^6 per foot respectively.

Surface temperatures and aerodynamic heating rates were obtained for one station and three radial positions on the conical portion of the blunted nose cone (Round A-40) and at three stations on each of two longitudinal rays on the sharp cone (Round A-41). Uncertainties in the data reduction invalidated the laminar results but the heating rates obtained in turbulent regions on the conical sections showed good agreement with the theoretical heating rates based upon Eckert's flat-plate reference enthalpy method. The surface finish was specified as 32 microinches (RMS) maximum.

27. Buglia, J. J., "Heat Transfer and Boundary-Layer Transition on a Highly-Polished Hemisphere-Cone in Free Flight at Mach Numbers up to 3.14 and Reynolds Numbers up to 24 x 10^6," NASA TN D-955, September 1961.

The model used for the flight-testing had a ratio of nose radius to base radius of 0.74 and a half-angle of 14.5° and was flight-tested at Mach numbers up to 4.70. The nose of the model had a surface roughness of 2 to 5 microinches as measured with an interferometer. A comparison with results from rougher models indicated that the high degree of polish was instrumental in delaying the transition from laminar to turbulent flow. However, data were obtained at Mach numbers of 2.32, 2.47, 2.63, 2.80, 2.97, and 3.14 and probably represent the most complete set yet obtained for correlation with theoretical methods. Stagnation point heating values were in best agreement with the prediction of Cohen while Lees theory best represented the distribution of heat transfer. The turbulent flow over the conical portion of the vehicle correlates best with the Eckert flat-plate reference enthalpy method.

28. Carter, H. S. and Bressette, W. E., "Heat Transfer and Pressure Distribution on Six Blunt Noses at Mach Number of 2," NACA RM L57C18, April 1957.

The heat transfer and pressures on the surfaces of six blunt-nose models are presented for 0° and 5° angles of attack. The tests were made in a free jet at a Mach number of 2 with a total temperature of 935°R and a Reynolds number per foot of about 14×10^6. The authors state that the heat transfer over the conical portion of the models was correlated with Van Driest's flat-plate theory while the heat transfer at the stagnation point was compared with the theory of Reshotko and Cohen. However, the data does not compare well with the results of Van Driest when comparison is based upon local conditions at the outer edge of the boundary layer. Insufficient data precluded comparison on the reference-enthalpy method.

29. Carter, H. S. and Carr, R. E., "Free-Flight Investigation of Heat Transfer to an Unswept Cylinder Subjected to an Incident Shock and Flow Interference from an Upstream Body at Mach Numbers up to 5.50," NASA TN D-988, October 1961.

Skin-temperature measurements were made along the stagnation line of the transverse cylinder of a cruciform test configuration constructed of two nearly equal size tubes. The flow on this stagnation line was influenced by boundary-layer separation and by end effects as well as by the intersecting shocks. Under these conditions the measured heat-transfer rates had inconsistent variations both in magnitude and distribution which precluded correlation of the heating rates with position of the intersecting shocks or with theory.

30. Chauvin, L. T., "Pressure Distribution and Pressure Drag for a Hemispherical Nose at Mach Numbers 2.05, 2.54, and 3.04," NACA RM L52K06, December 1952.

The results of an experimental investigation of the pressure distributions on a hemispherical nose 3.98 inches in diameter, mounted on a cylindrical support, were presented for wind tunnel Mach numbers of 2.05, 2.54, and 3.04 and for Reynolds numbers of 4.44×10^6, 4.57×10^6, and 4.16×10^6, respectively, based on body diameter and free-stream conditions.

This report presents basic tunnel operating parameters required for the evaluation of the convective heat transfer results presented by Chauvin and Maloney in NACA RM L53L08a (February 1954).

31. Chauvin, L. T. and Maloney, J. P., "Experimental Convective Heat Transfer to a 4-Inch and 6-Inch Hemisphere at Mach Numbers from 1.62 to 3.04," NACA RM L53L08a, February 1954.

Equilibrium skin temperatures and convective heat transfer were measured for a 4-inch and 4-inch diameter hemispherical nose in the Langley 9-inch and 12-inch blowdown wind tunnels, respectively. The Mach number for the 4-inch nose ranged from 1.62 to 3.04 for a Reynolds number based on body diameter of approximately 4.5 x 10^6. The 6-inch nose was tested at M = 1.99 at a Reynolds number based on body diameter of 6.4 x 10^6 and for two surface conditions.

The measured Stanton numbers correlated well with the Korobkin equation over the spherical section of the hemisphere for the laminar region. In the turbulent region the flat-plate reference method of Eckert represented the data fairly well. Transition was observed at Reynolds number of approximately 1 x 10^6 based on distance from the stagnation point and conditions just outside the boundary layer and corresponding to a region between the 45° and the 60° stations.

32. Cohen, N. B., "Boundary-Layer Similar Solutions and Correlation Equation for Laminar Heat-Transfer Distribution in Equipibrium Air at Velocities up to 41,000 Feet Per Second," NASA TR R-118, 1961.

Exact solutions for flat-plate flow, axisymmetric stagnation flow and stagnation flow for a yawed infinite cylinder are obtained and correlating formulas for the appropriate heat transfer functions are developed. A correlation formula for the heat transfer distribution function for an arbitrary favorable pressure gradient on a body of revolution or yawed infinite cylinder is determined for application to various local similarity methods. Pertinent equations are collected in a single section for convenience.

The methods presented in this report should be extended to lower Mach numbers. The correlation presented agrees better with experimental data than other well-known formulas and the range of application should be extended.

33. Crawford, D. H. and McCauley, W. D., "Investigation of the Laminar Aerodynamic Heat-Transfer Characteristics of a Hemisphere Cylinder in the Langley 11-Inch Hypersonic Tunnel at a Mach Number of 6.8," NACA Report 1323, 1957.

Aerodynamic heat transfer data to a nonisothermal hemisphere cylinder evaluated in the Langley 11-inch hypersonic tunnel at a Mach number of 6.8 and a Reynolds number from approximately 0.14 x 10^6 to 1.06 x 10^6 based on diameter and free-stream conditions were reported. The experimental heat-transfer coefficients were slightly less over the whole body than those predicted by the theory of Stine and Wanlass (NACA TN 3344) for an isothermal surface. For stations within 45° of the stagnation point the heat-transfer coefficients could be correlated by a single relation between local Stanton number and local Reynolds number.

34. Cresci, R. J., MacKenzie, D. A. and Libby, P. A., "An Investigation of Laminar, Transitional, and Turbulent Heat Transfer on Blunt-Nosed Bodies in Hypersonic Flow," Journal of the Aerospace Sciences, Vol. 27, No. 6, June 1960, pp. 401-414.

Experimental results for laminar, transitional, and turbulent heating rates measured by means of the shrouded model technique were reported. The Reynolds number was varied over a range from 0.4×10^6 to 4.0×10^6 while the stagnation to wall enthalpy ratio from 2.3 to approximately 1.5.

The experimental data were compared to the laminar hypersonic boundary layer theory of Lees and shown to be in good agreement on the conical portion of the model. On the spherical portion of the model the data are approximately 20 per cent higher than the theoretical prediction, due apparently to radiation to the model.

The experimental data presented in this study is not directly applicable to the problem of determining the aerodynamic heat transfer to blunt bodies at lower supersonic Mach numbers. However, the method of analysis should be explored for application in the transitional flow.

Extensive data are also presented for comparison of the flat-plate reference-enthalpy theory for the prediction of turbulent heat transfer to blunt bodies. It should be noted, however, that these data were obtained for stagnation temperatures in the range of 1500°R to 1700°R which correspond to flight Mach numbers of 10 and above.

35. Diaconis, N. S., Wisniewski, R. J., and Jack, J. R., "Heat Transfer and Boundary-Layer Transition on Two Blunt Bodies at Mach Number 3.12," NACA TN 4099, October 1957.

Local heat transfer values were measured on a hemisphere cone cylinder of diameter of 1.4 inches and a 120° included angle cone cylinder at a Mach number of 3.12. Experimental values of the heat transfer distribution for the blunt body agreed well with the heat transfer distribution prediction of the theory of Lees when the surface roughness was less than 16 micro-inches. An insufficient amount of wall temperature information prohibited comparison of the distribution results.

36. Hermann, R. and Melnick, W. L., "Aerodynamic and Heat Transfer Studies with Evaporative Film Cooling at Hypersonic Mach Numbers," RAL RR 189, University of Minnesota, Rosemount, Minnesota, Rosemount Aeronautical Laboratories, September 1962.

In order to establish the insulating characteristics of an evaporating film, measurements of the aerodynamic heat transfer to a hemispherically-capped cylinder were made at free-stream Mach numbers of 0.30 and 6.8 without mass injection. The low speed results ($M_\infty = 0.30$) indicated that the heat transfer conductance (h) were constant in the stagnation region, in agreement with well-known theories (where $Nu_R = 0.93\, Re_R^{1/2} Pr^{0.4}$). The laminar heat transfer distribution on the hemisphere cylinder at $M_\infty = 6.8$ with a Reynolds number of 283,000 per foot was well represented by the theory of Kemp, Rose, and Detra.

37. Kemp, N. H., Rose, P. H., and Detra, R. W., "Laminar Heat Transfer Around Blunt Bodies in Dissociated Air," Journal of the Aerospace Sciences, Vol. 26, No. 7, July 1959, pp. 421-430.

An extension of the method of Fay and Riddell is presented for predicting laminar heat-transfer rates to blunt, highly-cooled bodies with constant wall temperature in dissociated air flow.

Experimental heat-transfer rates obtained from shock tube facilities were presented and compared with the theoretical results. Approximately 75 data points were presented for the heat transfer distribution over a hemispherically-capped cylinder. These data were obtained in a shock tube at shock Mach numbers from approximately 8 to approximately 14, and hence the resulting experimental values of stagnation point heat transfer and heat transfer distribution are not applicable for flight Mach numbers less than 4.

38. Korobkin, I., "Local Flow Conditions, Recovery Factors and Heat-Transfer Coefficients on the Nose of a Hemisphere-Cylinder at Mach Number of 2.8," NAVORD Report 2865, White Oak, Maryland, U.S. Naval Ordnance Laboratory, May 1953.

Recovery temperatures and heat transfer coefficients were measured on an one-inch diameter hemispherically capped cylinder in the NOL 40 x 40 cm Aeroballistics Tunnel No. 3 at a Mach number of 2.8. The local heat-transfer coefficient for supersonic flow over the hemisphere with a laminar boundary layer were correlated with the incompressible expression presented by Sibulkin:

$$N_{u_D} = K(\beta)\, R_{e_D}^{0.5}\, Pr^{0.4}$$

39. Korobkin, I., "Laminar Heat Transfer Characteristics of a Hemisphere for a Mach Number Range 1.9 to 4.9," NAVORD Report 3841, White Oak, Maryland, U.S. Naval Ordnance Laboratory, October 1954.

Pressure and heat transfer measurements were made on the hemispherical nose of a two-inch diameter hemisphere cylinder with laminar boundary layer flow. The tests were conducted in the NOL 40 x 40 cm Aeroballistics Tunnel No. 1. Heat transfer measurements were made in the Mach number range from 1.90 to 4.87. The resultant Stanton number correlates with the expression for the stagnation region of a sphere

$$C_H = 0.763\, (Pr)^{-0.6}\, (Re_s)^{-0.5}$$

where Pr = 0.70 and the Reynolds number is based upon the local conditions at the outer edge of the boundary layer and the length parameter is the wetted surface distance from the stagnation point. These data are presented in Figure 19.

40. Marvin, J. G. and Deiwert, G. S., "Convective Heat Transfer in Planetary Gases," NASA TR R-224, July 1965.

Equilibrium convective heat transfer in several real gases was investigated. The gas considered were air, nitrogen, hydrogen, carbon dioxide, and argon. Solutions to the similar form of the boundary-layer equations were obtained for flight velocities from 10,000 to 30,000 ft/sec for a range of parameters sufficient to define the effects of pressure level, pressure gradient, boundary-layer edge velocity and wall temperature. Results are presented for stagnation point heating and for heating-rate distribution.

Although the range of velocities considered in this reference are beyond those of this bibliography, the method of analysis and trends indicated may be useful in developing methods of numerical solution of lower Mach number heat transfer problems.

41. Nardo, S. V. and Sadler, R. W., "Heat Transfer and Temperature Distribution in a Hemispherical Nose Cone in Hypersonic Flow," PIBAL Report No. 597, Polytechnic Institute of Brooklyn, New York, Brooklyn, New York, January 1962.

The heat transfer to a hemispherical nose cone subjected to hypersonic flow conditions was calculated by the theory of Lees for Reynolds numbers from one to five million. Comparison with results of test runs in the Polytechnic hypersonic tunnel facility indicated good agreement. The experimental procedure employed the shrouded model technique which is described in the paper of Cresci, MacKenzie, and Libby (Reference 34).

This report presents further details of a computational procedure for determining the heat transfer in the transition region from laminar to turbulent boundary layers on a hemispherical nose. Further experimental verification of the procedure is also presented.

42. Neal, L., Jr. and Bertram, M. H., "Turbulent Skin Friction and Heat Transfer Charts Adapted from the Spalding and Chi Method," NASA TN D-3969, May 1967.

Charts are presented which allow a rapid determination of local and average skin friction and heat transfer on flat plates in air. The charts cover a Mach number range from 0 to 20, a Reynolds number range from 10^5 to 10^9 in decade increments of the exponent and a wall temperature stagnation temperature ratio range of 0.1 to the adiabatic wall case.

The accuracy of the charts has not been established in the Mach number range from 0.8 to 5 since the experimental heat transfer data used to support the method employed in the development of the charts was obtained in the Mach number range from about 5 to 9.

43. Sands, N. and Jack, J. R., "Preliminary Heat Transfer Studies on Two Bodies of Revolution at Angle of Attack at a Mach Number of 3.12," NACA TN 4378, September 1958.

The axial temperature and Stanton number distributions for two pointed bodies of revolution at zero angle of attack are presented. No comparisons were made to theory.

44. Schurmann, E. E. H., "Engineering Methods for the Analysis of Aerodynamic Heating," RAD-TM-63-68, Wilmington, Massachusetts, Research and Advanced Development Division, Avco Corporation, November 1963.

Various methods of analysis of the convective heating to a body are reviewed for flight Mach numbers above 5. This report is of value in comparing methods of analysis with experiment for Mach numbers from 8.5 to 16 but does not contribute to evaluation of heating effects below Mach numbers of 5.

45. Seiff, A., "Examination of the Existing Data on the Heat Transfer of Turbulent Boundary Layers at Supersonic Speeds from the Point of View of Reynolds Analogy," NACA TN 3284, August 1954.

Heat transfer data from four wind tunnel experiments and two free-flight experiments with turbulent boundary layers were compared with Reynolds analogy and agreement was found within eight per cent. The data covered the Mach number range from 1.4 to 3.2 at effective Reynolds numbers from 0.4 million to 24 million. They were obtained on a variety of body shapes including flat plates, cones, and pointed slender bodies of revolution. Extremely blunt shapes such as spheres were excluded from the correlation.

46. Sommer, S. C. and Short, B. J., "Free Flight Measurements of Turbulent-Boundary Layer Skin Friction in the Presence of Severe Aerodynamic Heating at Mach Numbers from 2.8 to 7.0," NACA TN 3391, 1955.

Experimental measurements of average skin friction of the turbulent boundary layer were presented for a free-flying, hollow-cylinder model at Mach numbers of 2.8, 3.8, 5.6, and 7.0 at conditions of high rates of heat transfer. No heat transfer data or correlations are presented.

47. Stine, H. A. and Wanlass, K., "Theoretical and Experimental Investigation of Aerodynamic Heating and Isothermal Heat Transfer Parameters on a Hemispherical Nose with Laminar Boundary Layer at Supersonic Mach Numbers," NACA TN 3344, December 1954.

Measurements of the heat-transfer parameter $Nu/\sqrt{R_e}$ based on local flow conditions just outside the boundary layer and length of boundary-layer run were carried out on the heated, isothermal surface of a hemisphere cylinder for a Mach number of 1.97 and Reynolds number based on body diameter from 0.60×10^6 to 2.28×10^6. An approximate method for calculating the distribution of the heat-transfer parameter, applicable to any body of revolution was developed which predicts within about $\pm$ 18 per cent the experimental results. Since the data presented in the above study were obtained from an internally heated body, they were not subjected to analysis and correlation in our study.

48. Swanson, A. G., Buglia, J. J., and Chauvin, L. T., "Flight Measurements of Boundary-Layer Temperature Profiles on a Body of Revolution (NACA RM-10) at Mach Numbers from 1.2 to 3.5," NACA TN 4061, July 1957.

A parabolic body of revolution with a pointed nose, the NACA RM-10 was flight tested at Mach numbers up to 3.5 and Reynolds numbers up to 140×10^6 (based on length to the probe and rake measuring station). Skin-friction coefficients determined by the momentum method from the total-pressure rake data were in fair agreement with the Van Driest theory for flat plates with compressible turbulent boundary layers. Heat transfer measurements indicated a transition Reynolds number of about 15×10^6.

49. Wilson, R. E., "Viscosity and Heat Transfer Effects," (Sections 13 and 14, Handbook of Supersonic Aerodynamics), NAVORD Report 1488, Vol. 5, August 1966.

This report presented a review and discussion of nearly two hundred papers pertaining to the study of viscous effects in high-speed flows. The emphasis in this volume is placed on the presentation of theoretical and experimental results which supposedly can be used by the missile designer for the calculation of skin friction and heat transfer.

The method of calculation suggested in this report for the heat transfer to blunt bodies of revolution is, however, incomplete in this discussion of stagnation point heat transfer and the distribution of heat transfer over the body surface. Results are presented for high Mach numbers but the report does not extend the analysis to Mach numbers less than 5 but greater than 0.87.

50. Winkler, E. M. and Danberg, J. E., "Heat Transfer Characteristics of a Hemisphere Cylinder at Hypersonic Mach Numbers," NAVORD Report 4259, White Oak, Maryland, U.S. Naval Ordnance Laboratory, April 11, 1957.

The heat-transfer characteristics of the laminar compressible boundary layer on a hemisphere cylinder were presented for free-stream Mach numbers of 5, 6.5, and 8. These data have not been included in this bibliography. However, it should be noted that excellent correlation was attained with Van Driest's stagnation region expression for incompressible flow.

51. Zoby, E. V. and Sullivan, E. M., "Correlation of Free-Flight Turbulent Heat Transfer Data from Axisymmetric Bodies with Compressible Flat-Plate Relationships," NASA TN D-3802, January 1967.

Published experimental turbulent heat transfer data obtained over a range of free-flight conditions and body shapes (free-stream Mach numbers from 2.9 to 13.4, free-stream Reynolds numbers per foot from 0.64×10^6 to 30.7×10^6, body shapes from a hemisphere cylinder to a sphere cone with a half angle of 25°) were compared with calculated turbulent flat-plate values. The calculated values were evaluated by use of a modified Reynolds analogy and the skin-friction relationships of Blasius or Schultz-Grunow with compressibility effects accounted for by evaluating the flow properties on reference conditions. For reference Reynolds numbers less than 10^7, the calculated heating rates based on either of the two methods correlated well with the experimental data. For reference Reynolds numbers greater than 10^7 and less than 8×10^7, the calculated heating rates based on the Schultz-Grunow relation compare better with the available experimental data.

Zoby and Sullivan point out that available flight-test data correlate within 20 per cent of the correlation line whereas wind tunnel data spread out to ± 50 per cent of the correlation line. This point should be pursued in further study.

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13. ABSTRACT

A literature survey and an evaluation of applicable literature concerning pressure and heat-transfer distributions over bodies of revolution in high-subsonic to low-supersonic flight ~~is~~ are presented. Methods for determining the pressure distribution on bodies of revolution are discussed and compared with experimental data. Variations of the modified Newtonian method may be applied to a large class of bodies of revolution for free-stream Mach numbers greater than 0.75.

Appropriate experimental laminar and turbulent heat-transfer data for a blunt-body vehicle at flight Mach numbers between 1.6 and 5 are compared with calculated values. Experimental data could not be found for blunt-body turbulent heat transfer in the free-stream Mach number range from 0.7 to 1.6.

An extensive bibliography covering prediction techniques and experimental data on pressure and heat-transfer distributions is included as the appendix. ( )

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